SPACE TRANSPORT CAPABILITIES OF CHEMICALLY-FUELED PROPULSION SYSTEMS USING STORABLE AND CRYOGENIC PROPELLANTS

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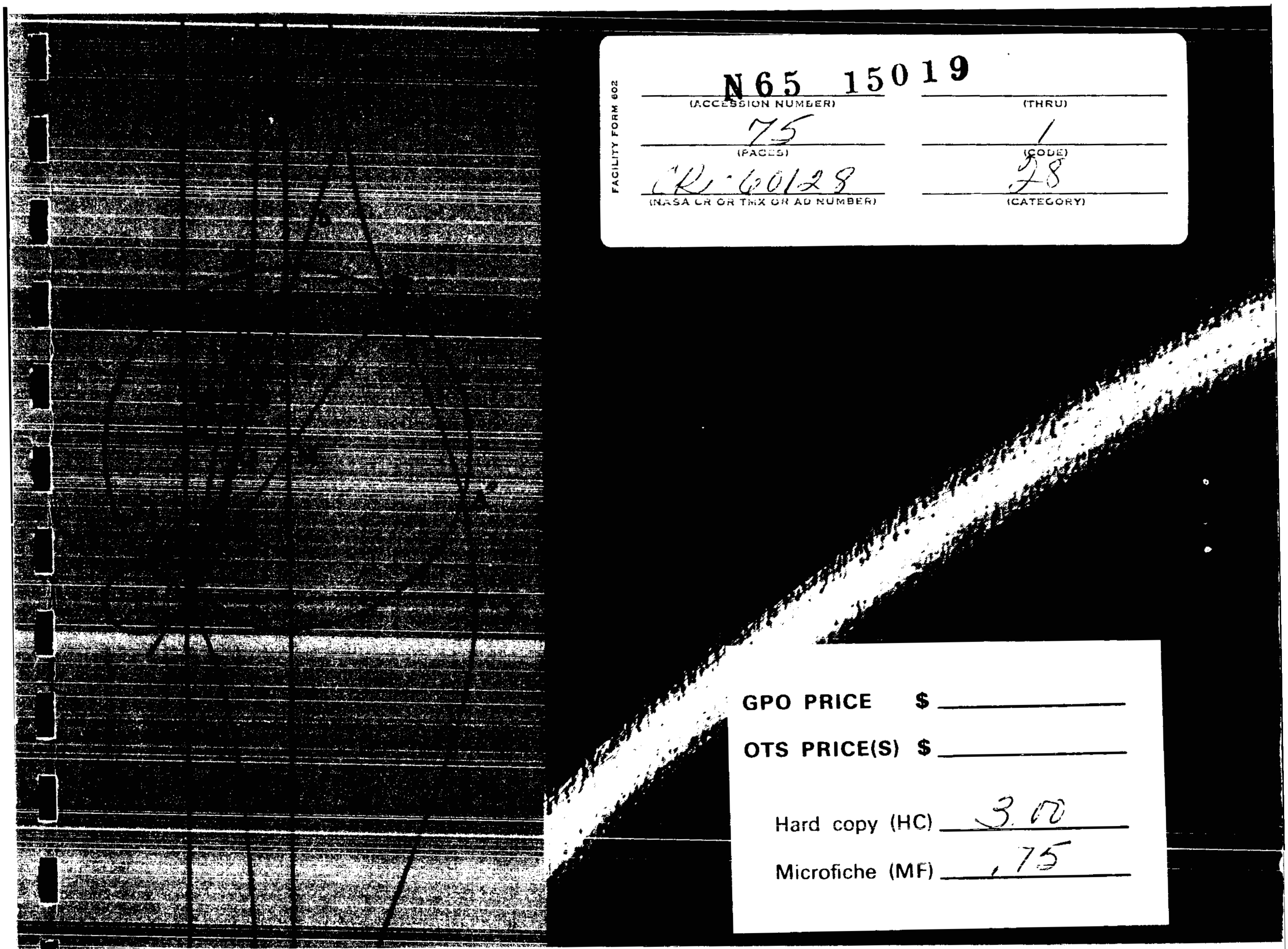

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SPACE TRANSPORT CAPABILITIES OF CHEMICALLY-FUELED PROPULSION SYSTEMS USING STORABLE AND CRYOGENIC PROPELLANTS

by

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Summary Report

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<u>ABSTRACT</u>

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This report documents the results of investigations into the relative transport capabilities of chemically-fueled upper stages using cryogenic and storable propellant combinations, these stages being designed to meet the same space mission objectives in a near optimum manner.

A terminal maneuver after a coast period characterizes the missions forming the basis of comparative evaluations. Various stage weights, propellant combinations, space storage methods, and thermal insulation systems are considered.

I. SUMMARY

A. PURPOSE AND SCOPE

The major purpose of this work is to utilize the most recent knowledge in cryogenic technology in application to upper stage space vehicles using cryogenic propellants in order to better define the transport capabilities of these vehicles compared with those using space storable propellants in a mission spectrum representative of NASA's future programs.

The major criterion for comparative evaluation is the deliverable payload. A terminal maneuver after a coast period characterizes the missions considered. The mission spectrum of interest include the lunar, solar, Mercury, Venus, Mars, and Jupiter probe. The liquid propellant combinations considered are:

1) $H_2 - O_2$
2) $H_2 - F_2$
3) $CH_4 - OF_2$
4) $H_2 - OF_2$
5) $N_2H_4 - N_2O_4$
6) $A50 - N_2O_4$
7) $B_2H_6 - OF_2$

The size of the stages considered in our evaluation range from 6000 to 40,000 pounds at departure from earth orbit. The weight penalties associated with various space storage methods and thermal insulation systems for the cryogenic propellants as based on the state-of-the-art technology and foreseeable extensions of that art are investigated as part of this study. In addition, special operational and technological problems associated with the application of specific propellant combinations are discussed.

B. CONCLUSIONS

The transport capabilities of upper stage space vehicles using high energy cryogenic propellants should exceed those using earth storable propellants in a number of missions within the spectrum of NASA's interest. The weight penalties imposed by the need for thermal protection of the cryogenic tankage due losses through boil-off (if any) depend on the mission requirement and on the propellant combination, but in a preponderance of cases investigated they are not so great as to obviate the basic payload advantage attendant to the use of high specific impulse propellant combinations.

In all cases investigated, the use of hydrogen-fluorine or diborane-oxygen difluoride resulted in the largest payloads, and, with very few exceptions, where one promised the greater transport capability the other was next in rank order. The superior specific impulse given by the hydrogen-fluorine propellant combination is responsible for its position. An excellent specific impulse, a relatively high density storage, and a relatively good space storability are qualities which account for the promise shown by the diborane-oxygen difluoride combination.

In the greatest majority of cases investigated, use of a fully mixed, non-vented space storage of the cryogenic propellants led to a greater deliverable payload than resulted from vented storage. For this reason, the trade-offs between increased payload and the more cumbersome and expensive ground-handling equipments and procedures and necessary mixing devices associated with the non-vented storage method deserves further attention.

No substantive investigation of the problems inherent to the handling of hydrogen-fluorine or diborane-oxygen difluoride propellants has been made. We note the technology of handling hydrogen-fluorine is much further advanced, but fuller knowledge of the handling problems are required in the interpretation of the results presented.

There is no current practice for thermally protecting space-borne cryogenic propellant tankage. The methods of thermal protection and the weight penalties associated with them that are factored into this study are based on current developments in this area and reflect their logical culmination.

II. METHODS

A. BASIS

The present study is aimed at determining, for various propellant combinations, the payload mass delivered in various missions by an upper stage having a given mass at earth escape. The payload mass is herein defined as that remaining after the masses of components necessary for the mission are deducted from the earth escape mass (or gross mass) of the upper stage vehicle.

The mission begins at earth escape, continues through the coast period, and ends in a terminal maneuver. The terminal maneuver requires propellant and the associated hardware: tankage, engine assembly, pressurization and expulsion systems, and thrust structure. The thermal environment during groundhold, earth ascent and coast impose the need of thermal protection for propellant tankage, particularly in the case of cryogenic fluids. Earth ascent thrust and moments impose structural requirements that are more severe than those associated with terminal thrust, and usually control in the design of the upper stage structure.

For the purpose of the study, the mission is typified by: a coast period, τ_o; a time integral of solar flux intensity, $I\tau_o$; and a terminal velocity increment, ΔV.

The vehicle is typified by its earth escape mass, or gross mass, M_G, and its configuration. The configuration used consistently as a reference in our study is shown in Figure 1.

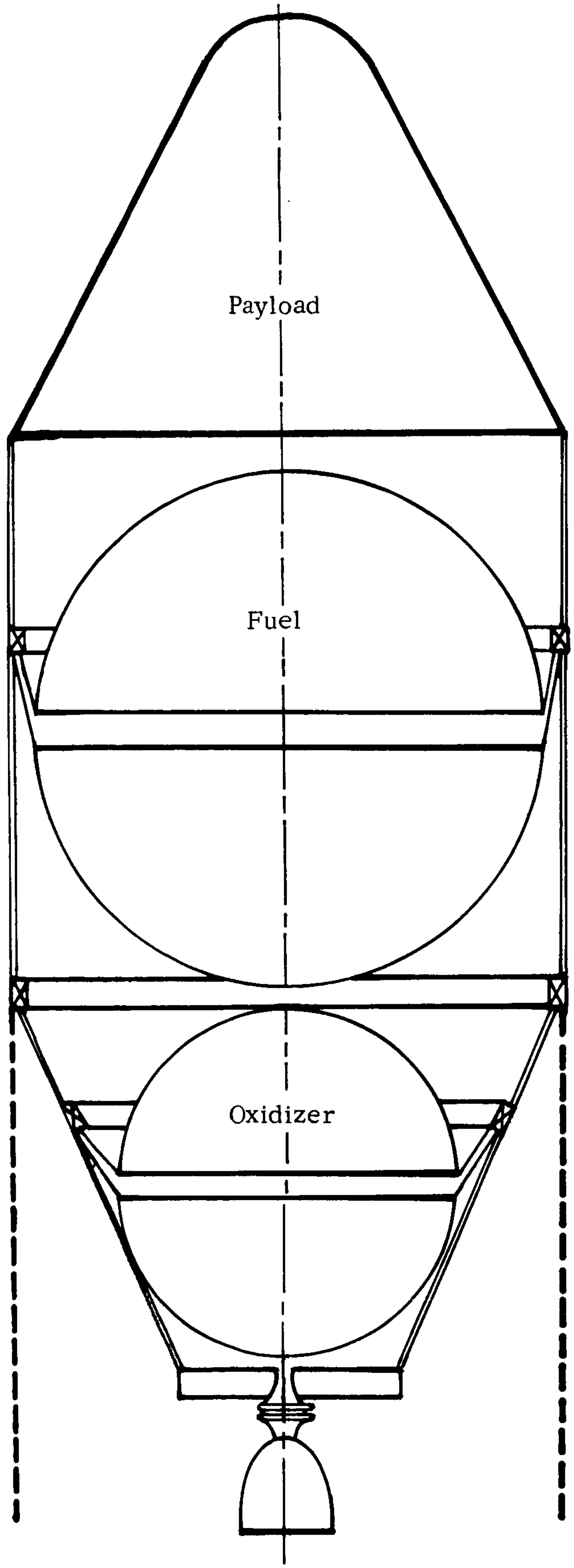

FIGURE 1 UPPER STAGE CONFIGURATION

Since the effect of choice of propellant combination is to be investigated, the propellant cannot be characterized realistically by one or two simple parameters. Rather, each propellant combination must be considered in terms of its own properties (optimum oxidizer-to-fuel ratio O/F and the corresponding specific impulse I_{sp}, relationship between density and other thermodynamic properties, normal boiling point, critical temperature, etc., for the fuel and for the oxidizer), which enter at different points in the analysis.

1. General Method

The general method applied in this study is as follows. First, a set of parameters defining a mission and a gross mass is chosen: ΔV, τ_o, $I\tau_o$, M_G. Next, the payload capability corresponding to this set is determined for each propellant combination listed in Section IA.

From ΔV, I_{sp} and M_G, the mass of useful propellant, M_P, required for the terminal maneuver is determined. From the O/F ratio, the useful masses of fuel, M_F, and of oxidizer, M_{OX}, are found.

From M_F, M_{OX}, $I\tau_o$, τ_o, the thermodynamic properties of the fuel, and considerations leading to the best thermal protection system, the (spherical) tank diameters, D_F and D_{OX}, mass of insulation, M_{INS}, if any, and the boil-off losses, M_{BO}, if any, are calculated, as well as the mass of the tanks and expulsion system, M_{TX}. Also, the upper stage dimensions are approximated.

We have considered a terminal thrust-to-earth-weight ratio equal to unity. Thus, the thrust is determined from M_G. The engine weight,

M_E, is found to depend, to first order, only on thrust, hence, only on M_G.

From I_{sp}, O/F, and the thrust, the mass flow rates of the fuel and oxidizer are found; in conjunction with the densities of these constituents, this permits a calculation of the mass of the turbopump assembly, M_{TPA}.

A maximum boost acceleration of 8 g, and a maximum lateral acceleration of 2 g, have been assumed consistently. This, together with the values of the component masses, their distribution in the stage, and the various dimensions found, allows the mass of the structure, M_{STR}, to be calculated.

Once the masses discussed above are found, the residual available mass is found by subtracting from M_G the sum of all the others: M_P, M_{INS}, M_{BO}, M_{TX}, M_E, M_{TPA}, and M_{STR}. This difference will be called the payload M_{PL}, and will, of course, include any electronic equipment, such as guidance.

2. Parametric Study

We have applied the general method just outlined to a number of missions defined by combinations of the parameters ΔV, $I \tau_o$, and M_G, wherein each of these was varied over an interesting range of values. For each set, the payload was calculated for each of the seven propellant combinations.

The parametric study has a three-fold purpose. First, it allows the coverage of a wide range of interesting cases. Second, it shows the effect of changes in one parameter with the others fixed. Third, it permits interpolation when a specific mission is being considered involving intermediate values of the parameters.

In this parametric study, only non-vented storage was considered. Also, no special account was taken of heat inleakage through certain small fixed conductive paths and heat sources. These limitations were imposed only by restrictions to the scope and effort of the program. The ranges of the parameters have been selected to exclude most cases where the effect of the fixed heat leaks cannot be neglected (e.g., small vehicles sent on missions of long duration).

3. Study of Specific Missions

A study of six specific missions was made. Four of these involved capture in a 300-nautical-mile circular orbit around the planets Mercury, Venus, Mars, and Jupiter. One mission is a solar probe involving transfer to a permanent circular orbit around the sun at a radius of 0.3 astronomical unit. The other mission involves a direct landing on the moon.

In the study of these missions, both vented and non-vented cryogenic storage were considered. Account was also taken of all sources of heat inleakage.

For each mission, three values of M_G were considered. The values of ΔV, $I\tau_o$, and τ_o were calculated for representative cases, i.e., those cited in contemporary analyses of unmanned missions, and projected for reasonable departure dates.

4. Special Missions

We have also considered two situations in which the manner of accounting for the stage components deviates from the ordinary case. The corresponding missions were chosen so that the deviations lead to the most significant changes in the resulting payload.

The first situation is that where the structure has been designed so that an appreciable fraction of it can be jettisoned just before the terminal maneuver. This results in reduced propellant requirements and accompanying changes in the masses of tankage, structure, etc. At a fixed M_G, the result is an increase in payload.

The second situation is that in which the stage escapes from earth after having been active previously, so that the tanks are only partly full, (or, conversely, the tanks are much larger than necessary). This results in a larger vehicle, with increased masses of tanks, insulation, structure, etc., and a consequent decrease in payload at fixed M_G. The overall effect can be fully assessed only if account is taken of the performance of previous stages, which is beyond the scope of the present study.

B. PROPERTIES

Table I lists some physical properties of the propellants that are introduced into our evaluations to an accuracy adequate for our purposes.

Figure 2 illustrates the internal energy change of the saturated liquid cryogenic propellants vs. pressure. These relationships are used in the determination of insulation requirements in cases of non-vented cryogenic storage as discussed in a succeeding section. The portion of the curves shown dotted indicate extrapolation of available measured data.

TABLE I

SOME PHYSICAL PROPERTIES OF PROPELLANTS

Propellant	Triple Point Temperature (^oR)	Normal Boiling Point (^oR)	Latent Heat of Vaporization at 1 atm (Btu/lb)	Density at Saturation Pressure 25 psi (lb/ft^3)	50 psi	100 psi
H_2	14	37	195	4.3	4.0	3.6
F_2	96	153	72	92	89	86
O_2	98	162	92	70	67.5	64
CH_4	163	201	218	26	25	23.7
OF_2	89*	231	88	94	92	89
B_2H_6	194*	325	224	23.3	22.2	21
N_2O_4	480*	530	-	86	84.5	81
A50	480*	630	-	52	50	47.5
N_2H_4	495*	696	-	55	54	53

* Freezing point at 1 atmosphere.

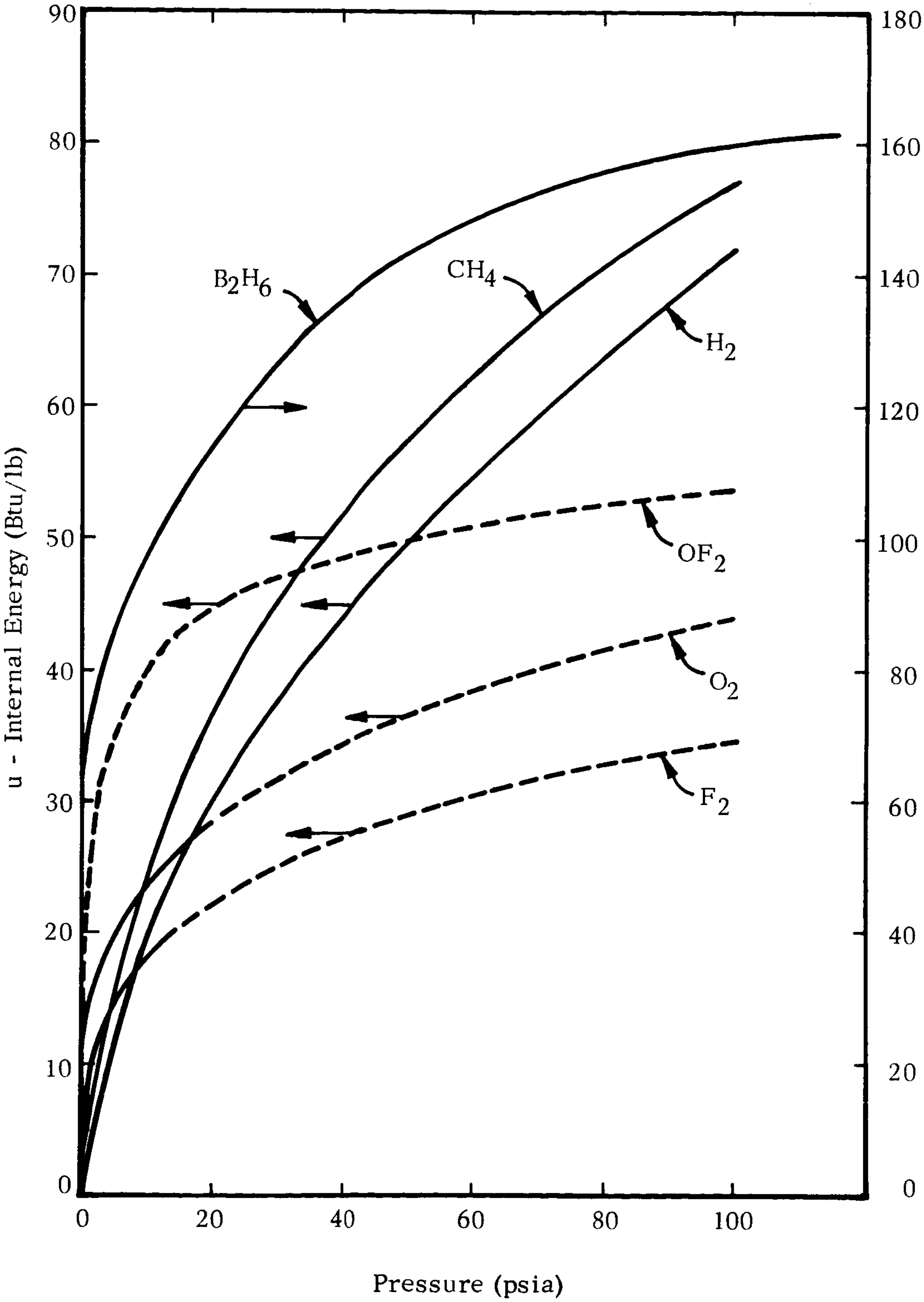

FIGURE 2 INTERNAL ENERGY OF SATURATED LIQUID CRYOGEN (u = 0 at Triple Point)

C. INSULATION REQUIREMENTS

The problem is to define an optimum thermal protection system for the cryogenic tankage that will limit the loss of cryogen after launch and that will withstand all the rigors of environment during the entire mission profile. The optimum thermal protection system would be one which will perform reliably and introduce a minimum weight penalty.

The weight penalty associated with the thermal protection system includes the weight of all components necessary for thermal conditioning that are carried into space and the weight of unavailable cryogen that is lost through venting and outage. In general, there is an additional penalty which is the increase in weight of the tank and expulsion system and structure sized to handle the propellant fraction that is lost through venting and to carry the weight of insulation.

There are two basic classes of thermal insulation systems which are considered, hereinafter referred to as Class A and Class B systems. Both of these systems use a multifoil, evacuated, radiation-shield type of configuration to minimize heat leaks during the stay time in space. The Class A system makes use of a light weight vacuum type encapsulation (say a Mylar bag) which allows for the maintenance of an acceptable vacuum during ground-hold and boost-out. In such a system, as long as the vacuum integrity of the bag is preserved, the heat inleakage to the hydrogen vessel can be maintained within acceptable limits during these portions of the mission. On the other hand, the Class B system uses a plastic foam (cork, plastic honeycomb, or equivalent) to limit the heat leakage during ground-hold and boost-out. In the Class B system the multifoil insulation is applied directly on top of the foam and is purged with helium to prevent contamination with the condensible gas constituents of air, and this part of the system provides only a small margin of thermal protection until it becomes evacuated in space. The Class A system may be regarded as a least weight configuration, but the efficacy of a vacuum-tight vacuum bag is questionable.

In both these systems, the multifoil component of the insulation system is determined by the space storage requirement. Weight optimization of this component for a vented system depends on the mission. For the simple firing schedule required to satisfy the missions forming the basis of this study, its weight should be made very nearly equal to the loss of cryogen due heat inleakage through the multifoil blanket divided by mass of the vehicle before and after the terminal maneuver. To this figure we must add the weight of the plastic foam (or its equivalent) and the insulation retention system for the Class B system and the weight of the encapsulating and insulation retention systems for the Class A type.

We have specified a reasonable limit to heat inleakage during ground-hold of 100 Btu/hr-ft^2. To meet this requirement in the case of the Class B system, we assume a thickness of reinforced foam of 5 pound density directly to the tank wall. The thickness is regulated in the case of each cryogen to limit the heat inleakage to the 100 Btu/hr-ft^2 figure with an ambient of 540°R and an external heat transfer coefficient equal to 1 Btu/hr-ft^2-$^{\circ}$R. This foam layer is carried into space although it provides comparatively little thermal protection during space operations.

In the case of the Class A system, the evacuated multifoil insulation, even though loaded with a one atmosphere pressure at the ground, will limit the heat leak to values below 100 Btu/hr-ft^2.

During boost-out the multifoil insulation layers are restrained by supporting nets (we have used vinyl covered fiberglass nets for the purpose in ground based applications) to withstand "g" loads, vibrations, and decompression forces. A small fixed weight penalty per unit of tank wall area has been applied to account for this support requirement. Protection from aerodynamic loads during boost-out are provided by the external shroud of the propulsion module. Decompression forces attendant to the boost-out phase for the ground-purged Class B multifoil layers are limited by perforation and/or controlled evacuation of the space within the shroud. The temperature pulse resulting from aerodynamic heating of the shroud during the boost phase may require special thermal protection features to prevent over heating of the multifoil layers (1)*. Aluminized Mylar foil superinsulation has less tolerance in this regard than the aluminum foil types. Nevertheless, in either case, the thermal protective features, if any, that may be required should not introduce a weight penalty significant to our comparative analyses.

During the coast period in space the heat inleakage to the cryogenic propellant tanks is limited by the application of the multifoil radiation shield type of insulation and by the careful design of heat resistant paths introduced by penetrations through the multifoil blanket

* Numbers in parenthesis refer to references listed at end of report.

necessary for support, pipe connections, etc.

The heat inleakage through the multifoil system is calculated by treating it as a blanket with heat transport characteristics in a direction normal to the tank wall that are dominated by thermal radiation effects and parallel to the tank wall by solid conduction. The total heat influx in such a system reduces to the black body emission from the outermost shield of the multifoil layer divided by a shielding factor. This shielding factor depends on the thermal properties of the shield and spacer combination making up insulation, and is very nearly proportional to the number of shields. As a consequence, the weight of the multifoil insulation per unit of area is also proportional to the shielding factor. Values for the shielding factor and weight per unit shielding factor are established from test results on the best multifoil insulations presently available [2].

Irrespective of the fact that the temperature of the outermost shield varies widely from location to location (i.e., from the sunlit side to the shady side), it is valid to treat the outermost shield as an isothermal surface equal to the adiabatic wall temperature. The adiabatic wall temperature is computed from a heat balance applied to the outermost spherical shield; the balance being achieved between the absorbed radiant thermal energy from external sources and that re-radiated to the space environment. For the missions projected in this investigation, sunlight is the dominant external source; therefore, the total heat inleakage to the tank through the insulating blanket is made proportional to the time averaged solar intensity times the coast period. We assume the outermost shield is coated to have a ratio of solar absorptivity to emissivity at its operating temperature of 0.3.

In effect, we treat the cryogenic tanks as if they were exposed to the space environment; in fact, they are enclosed within a nearly cylindrical envelope formed by the external shroud, the payload and engine. From a heat inleakage standpoint this enclosure is partly helpful (the shroud) and partly harmful (the near room temperature conditioned payload and perhaps the engine) and their combined effect is assumed to cancel. Finally, we have degraded the thermal performance of the blanket by 20 percent from the ideal performance obtained from measured data on carefully prepared samples to account for seams and discontinuity made necessary by application.

The heat inleakage via solid conduction through penetrations is based on the analysis of "weak thermal shorts" described in Reference 3. In the case of a weak thermal short the interaction between the penetrations and surrounding multifoil insulation is small and total heat inleakage can be estimated quite accurately by superposition.

The heat leak due penetrations is calculated on the basis of solid conduction via the cryogenic tank support. This support is assumed to be made of titanium tension members one foot long with sufficient cross section to support the propellant tank under an 8 g load. The temperature at one terminal is the temperature of the stored cryogen; the temperature of the other terminal is the time-averaged temperature of the shroud. The time-averaged temperature of the shroud is determined by assuming an isothermal cylindrical surface having an absorptivity to emissivity ratio of 0.3 in heat balance with the sun shining normal to its axis and reradiating to the star-speckled sky. The heat inleakage calculated on the basis described above is then multiplied by four, by way of introducing a factor to account for uncertainties and additional heat leaks due other penetrations such as pipes. In this way, we introduce heat inleakage to storage which is proportional to the amount of propellant stored and independent of the amount of multifoil insulation which may be applied.

Finally, we introduce a fixed amount of heat leak to each cryogenic tank equal to 4 Btu/hr to account for such things as instrumentation and in order to insert a factor consistent with experience which indicates a practical limit for heat inleakage for cryogenic tanks of the size of interest to this study.

In the case of the earth storable propellants, we assume no weight penalty for thermal protective means although it is clear that measures must be incorporated which prevent these propellants from freezing in some instances or exceeding storage tank pressure limits in others.

D. SPACE STORAGE OF CRYOGENIC PROPELLANTS

The long-term storage of cryogenic propellants in space depends on the use of highly-effective thermal insulation systems and, in some cases, auxiliary refrigerators. The requirements for fool-proof operation will favor the use of passive thermal protection methods, but reliable refrigerators can be developed should the savings in weight accompanying their use be sufficient to warrant their application.

The designer of the thermal protection system for cryogenic propellant tankage has several methods for preserving the propellant for the required storage period and each must be evaluated for its suitability in the application of his concern. For the storage of quantities measured in hundreds of pounds or more, the basic options reduce to: 1) storage at low pressure in a vented tank; 2) storage at low, variable pressure in a non-vented tank; 3) combinations of 1) and 2); and 4) storage making use of auxiliary refrigerators. A plurality of restraints imposed by a particular mission may dictate the selection of one of these methods, but a most common criterion for the choice is minimum total system weight.

In the vented system the heat inleakage to the stored cryogen results in boil-off losses. These losses reduce the propellant available after a coast period and require a storage and expulsion system and rocket structure made somewhat larger to accommodate the propellant which is eventually lost through venting. The boil-off losses can be reduced by adding insulation to the propellant tanks but a weight penalty is associated with this insulation and a portion of the heat inleakage due to penetrations for structural supports, pipes, and instrumentation is relatively insensitive to the thickness of the tank insulating blanket. As might be imagined an optimum exists where the weight penalty associated with propellant loss and insulation is minimized. For the characteristic missions of this study and where the propellant loss can be controlled within tolerable limits, thereby making the use of cryogenic propellants feasible, it can be shown that the highly effective evacuated multifoil insulation should be applied in an amount nearly equal to the weight of that portion of the propellant lost due to heat inleakage exclusive of penetrations divided by the mass of the vehicle before and after the terminal maneuver. Our calculations pertaining to vented storage are based on insulating systems which conform to this optimum; further, they reflect the weight penalties, tankage, expulsion system, and structure which result from the loss of propellant from storage and the weight of the applied insulation. In all cases the cryogenic propellant is assumed to be stored in space under saturated conditions at 15 psia.

In a non-vented storage system the heat inleakage to the stored cryogen increases its internal energy. As shown in Figure 2, there is a rise in pressure within the storage tanks commensurate with the increase in internal energy. To the degree possible insulation is applied in amounts sufficient to retain the cryogen without exceeding the pressure limits of the storage container. By making the tank stronger (and heavier) a lesser weight of insulation is required to preserve the storage. Again, an optimum combination of tank wall thickness and insulation thickness exists for minimum overall weight penalty. However, for most of the cases of interest in this study, this optimum is academic for it projects tank wall thicknesses less than those which are practical from the standpoint of fabricating leak-tight vessels. Therefore, in this study a minimum wall thickness requirement is imposed (.010 inch of Titanium alloy 5AL-2.5SN) and the tank is allowed to pressurize to within 20 psi of a safe operating pressure. The 20 psi margin is for expulsion gas pressurization for turbopump NPSH. In calculating the safe operating pressure, advantage is taken of the increased strength properties of titanium at low temperatures. In other words the allowable stress limits are varied depending on the temperature of the stored cryogen. In addition, the density change accompanying the pressure increase during the storage period for each cryogen, as well as ullage and outage requirements, are accounted for in sizing the storage vessels.

The start condition for the non-vented space storage period has been assumed to be 10 Btu per pound above the triple point condition for each cryogen in order to provide a margin for heat inleakage during boost-out and earth orbit. Also, in calculating the allowable heat capacity of the stored cryogens, we have assumed a well stirred, isothermal fluid. To meet this requirement will probably require auxiliary stirrers installed within the storage tanks. The weight penalty for these should be small and has been neglected.

In comparing the weight penalties of non-vented vs. vented storage means we note that shorter storage periods and larger cryogenic tankage favor the non-vented means and vice versa. Actually where the use of the vented storage means alone shows to advantage, one can demonstrate that a period of non-vented storage followed by a period of vented storage results in a lesser weight penalty. Although combination storage would be appropriate to some of the cases investigated, the limited scope of our investigations prevented parametric investigation of combined storage means.

The use of a refrigerator can result in a least weight storage system for the preservation of cryogenic propellants in space for long periods. Where the use of a refrigerator becomes appropriate depends basically on the type and amount of stored propellant and the mission. Where appropriate, optimum combinations of insulation and refrigeration are to be used. An electric power source having a capacity measured in hundreds or thousands of watts and a radiator for heat rejection from the refrigerator must be on board to supply the refrigerator. Figure 3 shows estimated weights of a space-borne refrigeration system for recondensing selected cryogenic propellants. This figure results from studies we have made of this problem. Reference 4 is an example of one. The weight of the refrigerator system as illustrated includes the power supply, power conditioning equipment, and space radiator estimated to weigh a total of 0.1 pound per watt of power demand. Refrigerators to meet the requirements of long reliable operation, small size and weight, and high efficiency demanded of this application are not now available. However, developments in space-borne refrigerators in progress give promise of meeting these needs.

In general, refrigeration shows a weight advantage first in the preservation of liquid hydrogen over the other cryogenic propellants. As a rule of thumb, we may say that refrigeration of the main propellants can be considered when the storage period exceeds one year. This rule is very approximate; methods have been developed [5] whereby the potential advantages of applying refrigerators can be more precisely quantified. In this study, the parametric evaluation of the

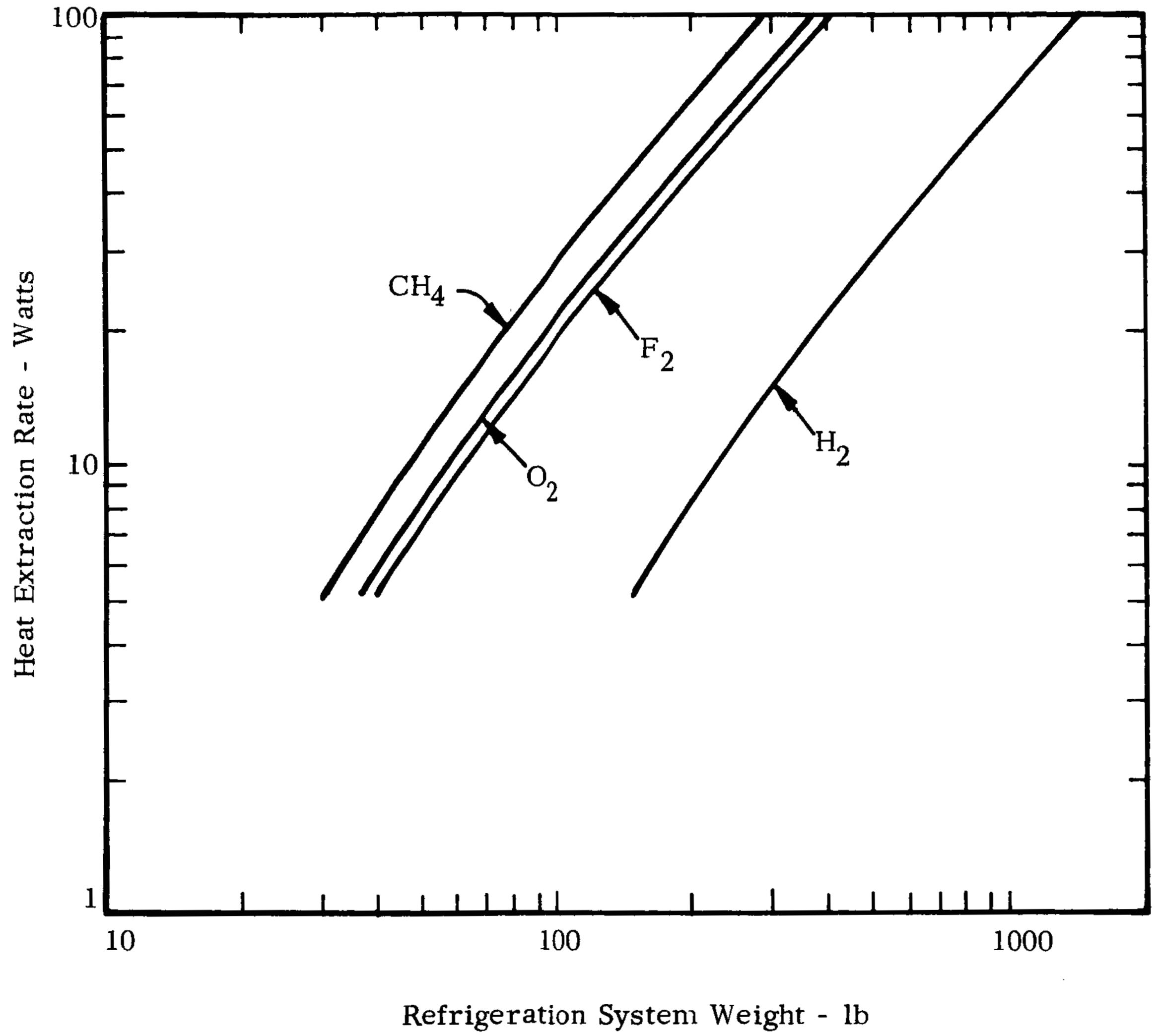

FIGURE 3 ESTIMATED WEIGHTS OF SPACEBORNE RECONDENSING SYSTEMS FOR CRYOGENIC PROPELLANTS

application of refrigerators has not been carried out. Rather, we can interpret the results of this investigation in the light of these prior studies and infer where the application of refrigerators should be considered.

E. STRUCTURAL REQUIREMENTS

The configuration of the stage that we have adopted as a basis for analysis was shown in Figure 1. Its main dimensions are determined by the tank diameters. The latter depend on the masses of the propellant constituents, M_F and M_{OX}. The loads applied to the structural members depend on the distribution of the masses. Therefore, once the various masses have been determined, it is possible to estimate the masses of the structural components and, hence, the total structural mass M_{STR}.

The manner in which M_{STR} is determined can be described with the use of Figure 4. The structure consists of an aluminum structural shell divided into three sections. Each of Sections I and II is cylindrical and of uniform strength throughout its length, and designed for the maximum bending load combined with the (constant) thrust load imposed on that Section. Section III is conical, but is assumed for analytical simplicity to be a cylinder having the same length as the cone but half the radius of the main shell.

The masses of the various structural sections are calculated using a procedure outlined by Sandorff [6]. For any cylindrical section under axial compressive thrust, T, a strength modulus requirement, $T/\pi R^2$, is calculated. A curve (Figure 2 in Sandorff) gives the product "equivalent shell thickness" times density over shell radius

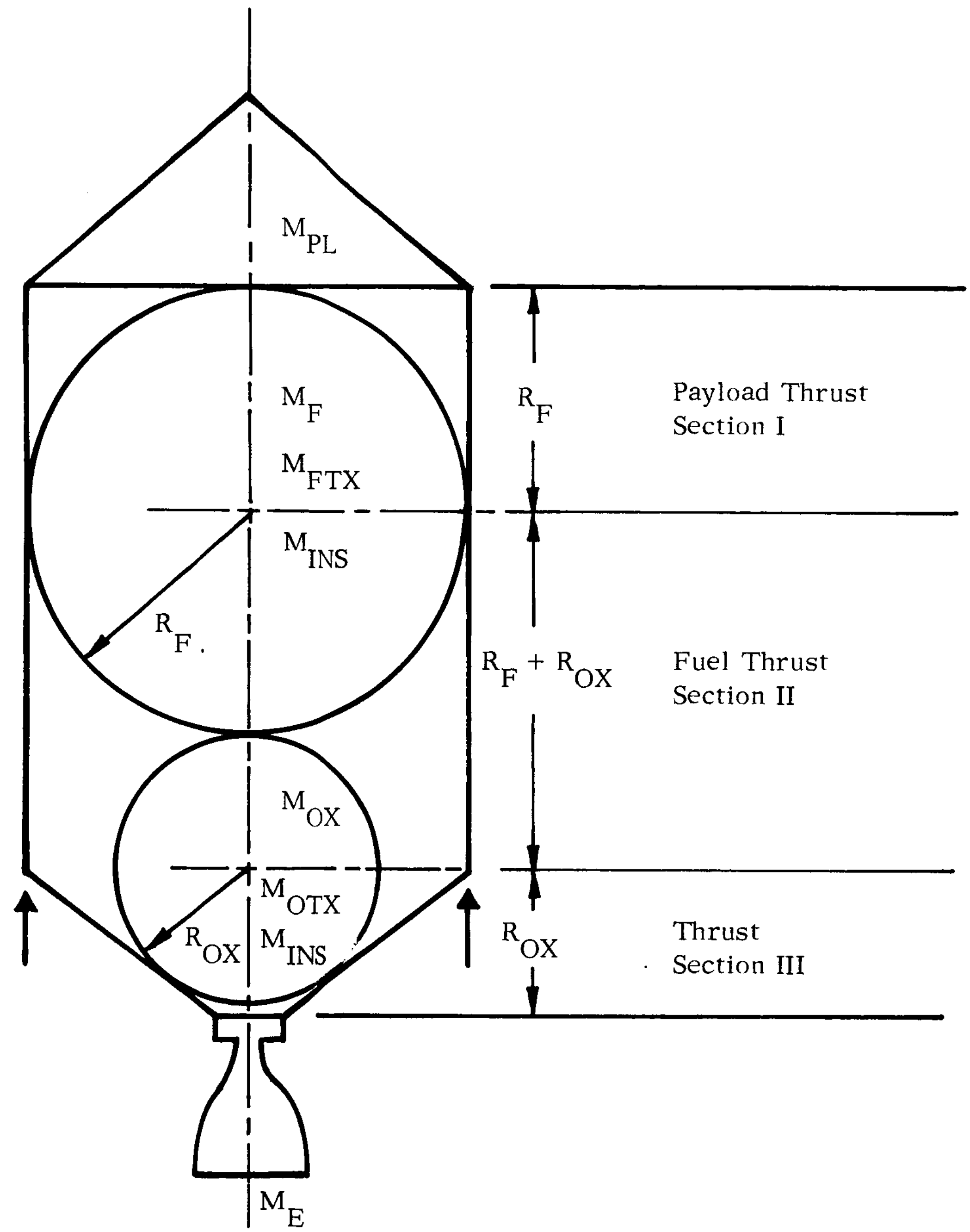

FIGURE 4 ANALYTICAL MODEL FOR STRUCTURAL ANALYSIS

as a function of the thrust modulus, for structural shells stiffened with stringers and rings. Since the length and radius of the shell are known, the mass of the section for thrust is easily obtained. A similar procedure (based on Figure 1 in Sandorff) can be followed, using a moment modulus $2M/\pi R^3$, to obtain the mass of the section to support a moment. The two masses are added. When the masses of all three sections are known, they are added to form a first estimate of structural mass. Finally, a 25 percent increase in this first estimate is added to account for local reinforcements and structural additions for tankage and expulsion systems support.

1. Loading During Boost

Section I must accelerate the payload to 8 g in the axial direction and give structural support to a 2 g lateral load (a thrust of 8 M_{PL} and a moment of 2 M_{PL} R_F).

Section II must accelerate the payload fuel tankage and expulsion system plus insulation and fuel (a thrust of 8 $(M_{PL} + M_{FTX} + M_{INS} + M_F)$ and a moment of $2 M_{PL} (2 R_F + R_{OX}) + 2 (M_{FTX} + M_{INS} + M_F)(R_F + R_{OX})$.

Section III is in tension and must accelerate the oxidizer, oxidizer tankage and expulsion system plus insulation, engine, and turbopump to 8 g. The moment requirements are small.

In all cases a lower limit of 0.025 inches was imposed on the "equivalent shell thicknesses" of the three sections to give some effect to practical minimums imposed by the stipulation of a continuous shroud and needs for fabrication and handling.

2. <u>Loading During Terminal Thrust</u>

For all cases considered, the thrust loads on Sections I and II can be shown to be highest at burn-out.

For Section II, this loading is always less than that during boost. Therefore, design for boost conditions automatically satisfies the requirements of terminal thrust.

For Section I the terminal thrust loading at burn-out is u M_{PL}, where u is the stage mass ratio (ratio of stage light-off to burn-out masses). This thrust can be greater than the maximum boost thrust load 8 M_{PL} since when low I_{sp} propellants are used for missions requiring high ΔV, u can be larger than 8. We have not taken account of this requirement, bearing in mind the following: (a) M_{STR} is of the order of 1.5 percent of M_G for the low I_{sp} propellants; (b) the mass of Section I is less than 30 percent of M_{STR}, hence, less than 0.5 percent of M_G; (c) these percentages decrease as u increases since the payload mass, which is supported by Section I, decreases with increasing values of u; (d) in all cases, a 25 percent weight factor has been added to the calculated value of the weight of structure, to account for reinforced sections; etc.; (e) the correction to M_{STR} that would result is within the possible error and is a refinement not warranted in the present study.

For Section III the terminal load is M_G in compression as compared with a boost tensile load of 8 $(M_{OX} + M_E + M_{TPA} + M_{OTX})$. Although the latter is from two to seven times the compressive load, Section III was designed for compression, because either (i) the shell thickness required by the tensile load is below the minimum thickness and is, therefore, unrealistic; or (ii) the compressive loading criterion imposes the greater weight penalty.

F. STORAGE TANKS

For purposes of weight estimation, we assume the storage tanks are spherical vessels designed as stressed membranes. The vessels are sized to retain the required amount of propellant to accomplish the mission in its least dense condition (a factor, although small, in the case of a non-vented storage) plus a 7 percent volume allowance for ullage and outage. The tanks are assumed to be made of 5 Al-2.5SN alloy of titanium and to have a wall thickness of 0.010 inches, which thickness is consistent with weight optimization. A 35 percent increase in weight over the constant wall thickness design is added to account for reinforcements for local stresses, internal piping and slosh baffles.

In the case of non-vented storage the maximum membrane stresses are reached at the end of the coast period. A thermal protection system is provided to limit the internal tank pressure plus an added allowance of 20 psi for gas pressurization to result in a design stress level that is 80 percent of the yield stress at operating temperature.

In the case of the vented storage, maximum stress levels are reached during boost-out and are below 80 percent of yield.

In those instances where the compatability of titanium with the propellant is questionable, for instance, its impact sensitivity with oxygen and fluorine, one can substitute a .025 inch thick wall of a high strength aluminum alloy such as 2014-T6 and fulfill the strength requirements at an increase in dry tankage weight of approximately 50 percent.

G. EXPULSION SYSTEM

The weight of the expulsion system is comprised mainly of the weight of the helium-filled expulsion gas bottles. We assume warm gas storage in titanium bottles. With equal stress limits for the propellant tanks and expulsion bottles, the weight of the expulsion gas bottles can be shown to be very nearly equal to the propellant tanks, and this equality is assumed in our evaluations. In addition the weight of expulsion gas is 15 percent of the expulsion gas bottles. The weight factor, M_{TX}, shown in the IBM printout sheet of results presented in Section K is the total of the weights of propellant tanks, expulsion gas bottles and expulsion gas in ratio 100:100:15.

H. THE ENGINE

Two approaches suggest themselves in estimating the weight M_E of the engine assembly (exclusive of the turbopump assembly). In the first approach, a design analysis is gone through; this analysis must be realistic and include all elements necessary to carry out the functions of the engine and to meet the mechanical strength and thermal (cooling) requirements. In the second approach, use is made of information on existing engine assemblies, with interpolation or extrapolation where necessary.

The design of a rocket engine requires attention to a considerable amount of detail, and, in our study, would have had to be repeated for a large number of cases. One item, the cooling system, involves a choice that depends on several quantities: heat transfer rate to the nozzle, burning time, chamber pressure. Although reliable ground work was already available [7], we chose not to adopt the design approach for the following reasons: (a) the state of the art relative to cooling methods is still not firm enough to permit of generalization; (b) the differences in M_E associated with different cooling methods are of the order of the error which can be tolerated in estimating M_E.

We neglect variations in the weights of the gas generator, combustion chamber, injectors and manifolds, and we assume cooling tube walls of fixed thickness (this actually minimizes the weight of a regenerative cooling system). As a result, it is possible to find a basic relationship between M_E and the product: (chamber pressure, P_c) x (throat area, A_t). This relationship has been plotted in a report by Aerojet-General [8], and is corroborated, in that points that represent existing engines fall close to the theoretical curves. We have adopted this relationship in our estimate of M_E.

The product $P_c A_t$ is equal, by definition, to the product of thrust (equal in our case to M_G) and the ratio: characteristic velocity c^* over specific impulse, I_{sp}. For all seven propellants considered, the ratio c^*/I_{sp} is found to vary by no more than 2.5 percent about a mean value of 0.524. Therefore, in the relationship suggested by Aerojet-General, M_E can be considered as depending only on M_G. Note that this dependence is not affected by a choice of P_c.

I. TURBOPUMP ASSEMBLY

A procedure similar to that used for the engine is applied to estimate the mass, M_{TPA}, of the turbopump assembly. We have considered separate turbopump systems for the fuel and the oxidizer, and added the mass of each to form M_{TPA}.

In the Aerojet-General report [9], the mass of turbopump systems is plotted against the ratio: mass flow rate over (propellant density)$^{0.8}$. The total mass flow rate of propellant is simply the thrust divided by I_{sp}; and again the thrust equals M_G. Therefore, the total mass flow rate can be found. Then, from a knowledge of the O/F ratio, the mass flow rate of each constituent is calculated. Using a mean density for the fuel and one for the oxidizer (employing the proper units as called for in reference 9), raising these values to the power 0.8, and dividing the results into the respective mass flow rates, one obtains values with which to determine the weight of the turbopump systems.

J. SPECIAL CASES

In this section we will consider two situations in which the events in a trip to space depart in some way from the standard sequence adopted throughout the remainder of the report. In the first of these situations, part of the structural shell is jettisoned just before the terminal maneuver. The second situation involves an upper stage that has escaped from earth with propellant tanks only partially full.

1. Jettisoning of Structure

From the discussion in Section E, Structural Requirements, it is clear that the mass of Section II of the structure is governed by the maximum thrust imposed during boost. In fact, the terminal load on Section II is so small compared to the boost load (plus moment) that it (the terminal load) could be transmitted by some light-weight internal strut arrangement. This suggests that a large fraction of Section II, which accounts for about 60 percent of M_{STR}, might be jettisoned before the terminal maneuver. This procedure will always lead to increased payload for a given M_G, but will produce a particularly significant effect, and, hence, will be most worthwhile, on missions involving a small payload fraction (large mass ratio u).

Suppose that a fraction α of the mass of structure, or αM_{STR}, is jettisoned just before the terminal maneuver. Then the light-off mass will be $M_G - \alpha M_{STR}$ instead of M_G. The required amount of propellant will be $(M_G - \alpha M_{STR})(1 - \frac{1}{u})$ instead of $M_G (1 - \frac{1}{u})$. Therefore, since the vehicle escapes from earth with a fixed mass M_G, and need carry less propellant, the gain can be transferred directly to increasing the payload. This increase is $\alpha M_{STR} (1 - 1/u)$. Of course, we have not considered changes in stage mass due to changes in the tank sizes, boil-off, insulation, etc.; these are of second order importance.

2. Earth Escape with a Fraction of the Propellant Previously Utilized

Consider a space vehicle entering an interplanetary orbit on a given mission, but with its tanks only partially filled with liquid. This situation could be the result of several possible circumstances, but in the present discussion the missing propellant is considered as having been used during the earth-escape maneuver.

The use of the upper stage, in lieu of the heavier second-to-last stage, to contain propellant for completing earth escape, will undoubtedly result in an increase in payload-to-launch weight ratio. However, since the determination of that effect is beyond the scope of the present analysis, we treat here only the performance of the upper stage; this information can later be used to evaluate total system performance.

As a basis for analysis, we consider an upper-stage vehicle designed for a given value of gross mass and a given mission, using a given propellant with vented storage. Specification of the mission and propellant implies the specification of the mass ratio u and the amount of propellant required for the mission with full tanks. Also implied are the masses of all elements in the vehicle, based on full tanks at earth escape; this includes the payload M_{PL} and M_{BO}.

If we now decrease the mass of propellant M_P at earth escape, M_{PL} must decrease. The relationship between these two masses is simple for the case of vented storage of propellants: the amount of boil-off is unchanged. The mass ratio u may then be expressed as unity plus the ratio $M_P/(M_{PL} + M_{FIXED})$, where the last term in brackets represents the total fixed mass of the vehicle elements. Since u is specified for a given mission, so is the above ratio; moreover its value is given (equal to u-1). This is the relation between M_P and M_{PL} that we shall use.

Since we are considering an upper stage vehicle designed for a given value of gross mass, and since all elements except the payload retain their respective masses before any of the propellant is used, the gross mass will be less than the design value by the decrease in M_{PL}. The vehicle mass at earth escape will equal the reduced gross mass less the amount of propellant used before earth escape. The ratio of these two quantities (reduced gross mass divided by the earth escape mass) is the mass ratio associated with the maneuver in which the missing propellant was used. This mass ratio, together with I_{sp} for the propellant in question, determines the velocity increment ΔV_e given the upper stage during the earth-escape maneuver.

K. RESULTS

1. Nomenclature

For convenience of interpretation, the nomenclature used in our evaluations as illustrated in the IBM printout of results is repeated.

ISP - specific impulse

ITO - average solar intensity times stay time in space

M BO - total mass of propellant lost due boil-off (vented storage)

M ENG - mass of engine

M INS - total mass of insulation on cryogenic propellant tanks

A - Class A

B - Class B

M PAY - mass of payload

M PU - total mass of propellant used in terminal maneuver

M PRO - total mass of propellant at escape from earth orbit

M STR - mass of support structure

M TPA - mass of propellant turbopump assemblies

M TX - total mass of propellant tankage and expulsion system

TO - stay time in space (coast period)

2. Parametric Evaluations

Tables IIA, IIB, and IIC summarize the results of parametric evaluations. The use of high energy cryogenic propellants show a payload advantage in all cases covered in the parametric matrix except possibly for the hydrogen-oxygen combination in small gross weight vehicles in missions with a terminal maneuver calling for a high velocity increment. The relatively high specific impulse of the hydrogen-

TABLE II A

PARAMETRIC EVALUATIONS

GROSS WEIGHT= 6000.LBS   M ENG= 73.LBS   NON-VENTED PROPELLANT STORAGE

PROPELLANT		M PRO	M TX	M TPA	M STR	M INS 3 A	M INS 3 B	M INS 30 A	M INS 30 B	M INS 100 A	M INS 100 B	M PAY 3 A	M PAY 3 B	M PAY 30 A	M PAY 30 B	M PAY 100 • A	M PAY 100 • B
VELOCITY INCREMENT= 8000. FPS																	
H2	O2	2590	127	22	153	14	60	19	65	32	78	3018	2973	3013	2968	3000	2955
H2	F2	2509	87	13	94	10	40	13	43	22	52	3212	3182	3209	3178	3200	3169
CH4	OF2	2728	55	9	52	6	22	6	22	8	23	3075	3059	3074	3059	3073	3058
H2	OF2	2547	102	16	121	12	48	16	53	26	63	3126	3089	3122	3085	3111	3075
N2H4	N2O4	3156	56	11	63	NO INSULATION							2639				
A 50	N2O4	3163	55	11	57	NO INSULATION							2640				
B2H6	OF2	2639	59	10	61	6	22	7	22	8	23	3150	3134	3150	3134	3149	3133
VELOCITY INCREMENT= 20000. FPS																	
H2	O2	4539	181	21	159	21	85	27	92	45	109	1003	939	997	932	979	915
H2	F2	4451	125	12	95	14	58	19	62	30	74	1227	1183	1222	1179	1211	1167
CH4	OF2	4682	78	9	52	9	31	9	32	11	33	1094	1072	1094	1071	1092	1070
H2	OF2	4492	146	16	125	17	69	22	75	36	89	1128	1076	1123	1070	1109	1056
N2H4	N2O4	5072	76	11	69	NO INSULATION							698				
A 50	N2O4	5077	75	11	59	NO INSULATION							703				
B2H6	OF2	4591	84	10	64	9	32	10	32	11	33	1166	1144	1166	1144	1165	1142
VELOCITY INCREMENT= 35000. FPS																	
H2	O2	5493	204	21	151	23	96	31	103	50	122	31	-41	24	-48	5	-67
H2	F2	5439	141	12	86	16	65	21	70	34	83	230	180	225	175	212	163
CH4	OF2	5577	87	9	48	10	35	10	35	12	37	193	168	193	168	191	166
H2	OF2	5465	166	16	115	19	79	25	85	41	100	144	85	138	79	123	63
N2H4	N2O4	5771	83	11	70	NO INSULATION							-8				
A 50	N2O4	5773	81	11	59	NO INSULATION							2				
B2H6	OF2	5524	95	10	61	11	36	11	36	12	38	224	198	223	198	222	197

• TEN THOUSANDS OF BTUS PER SQUARE FOOT

TABLE II B

PARAMETRIC EVALUATIONS

GROSS WEIGHT= 25000.LBS M ENG= 252.LBS NON-VENTED PROPELLANT STORAGE

PROPELLANT		M PRO	M TX	M TPA	M STR	M INS						M PAY					
					ITO=	3		30		100		3		30		100 *	
						A	B	A	B	A	B	A	B	A	B	A	B
VELOCITY INCREMENT= 8000. FPS																	
H2	O2	10793	312	88	772	36	147	46	157	73	183	12745	12634	12735	12624	12708	12597
H2	F2	10457	213	51	475	24	99	31	106	49	124	13524	13450	13517	13443	13499	13425
CH4	OF2	11369	139	38	265	16	55	17	56	19	59	12919	12879	12918	12878	12916	12876
H2	OF2	10613	253	65	612	29	120	37	128	59	150	13173	13082	13165	13074	13143	13052
N2H4	N2O4	13152	142	46	320	NO INSULATION							11085				
A 50	N2O4	13179	139	45	290	NO INSULATION							11093				
B2H6	OF2	10997	149	41	309	17	56	17	57	19	58	13232	13192	13231	13192	13230	13190
VELOCITY INCREMENT= 20000. FPS																	
H2	O2	18914	444	85	800	51	209	65	222	100	257	4451	4293	4437	4279	4402	4244
H2	F2	18547	306	50	484	35	142	44	151	68	175	5322	5216	5313	5207	5289	5183
CH4	OF2	19512	197	38	266	23	79	24	80	27	83	4710	4653	4708	4652	4706	4649
H2	OF2	18720	362	63	630	42	171	53	182	82	211	4928	4799	4917	4788	4888	4759
N2H4	N2O4	21135	195	46	331	NO INSULATION							3039				
A 50	N2O4	21156	190	45	289	NO INSULATION							3066				
B2H6	OF2	19130	214	40	324	24	81	25	81	27	83	5012	4955	5011	4955	5009	4953
VELOCITY INCREMENT= 35000. FPS																	
H2	O2	22890	501	85	647	58	235	73	250	111	289	565	387	550	372	511	333
H2	F2	22663	347	50	361	40	161	50	171	76	197	1285	1163	1274	1153	1248	1127
CH4	OF2	23240	221	37	214	25	88	26	90	30	93	1008	944	1006	943	1003	940
H2	OF2	22771	409	63	488	47	194	59	206	91	238	966	820	954	808	922	776
N2H4	N2O4	24047	212	46	313	NO INSULATION							128				
A 50	N2O4	24056	206	45	264	NO INSULATION							174				
B2H6	OF2	23020	242	40	270	27	91	28	92	30	94	1146	1082	1145	1081	1143	1079

* TEN THOUSANDS OF BTUS PER SQUARE FOOT

TABLE II C

PARAMETRIC EVALUATIONS

GROSS WEIGHT= 40000.LBS M ENG= 379.LBS NON-VENTED PROPELLANT STORAGE

PROPELLANT		M PRO	M TX	M TPA	M STR	M INS						M PAY					
					ITC=	3		30		100		3		30		100 *	
						A	B	A	B	A	B	A	B	A	B	A	B
VELOCITY INCREMENT= 8000. FPS																	
H2	O2	17269	419	137	1334	48	197	61	210	95	244	20409	20261	20396	20248	20363	20214
H2	F2	16731	287	81	823	33	133	42	142	64	164	21663	21563	21654	21554	21632	21532
CH4	OF2	18190	189	61	459	21	75	23	76	25	79	20698	20644	20697	20643	20694	20640
H2	OF2	16982	340	102	1059	39	161	50	171	77	199	21095	20974	21085	20963	21057	20936
N2H4	N2O4	21044	194	73	552	NO INSULATION							17755				
A 50	N2O4	21087	189	72	501	NO INSULATION							17770				
B2H6	OF2	17596	203	65	534	23	76	24	77	26	79	21197	21143	21196	21142	21194	21140
VELOCITY INCREMENT= 20000. FPS																	
H2	O2	30263	597	134	1393	69	280	86	298	130	342	7163	6951	7146	6934	7101	6890
H2	F2	29676	411	79	842	47	191	59	202	89	232	8562	8419	8551	8407	8520	8377
CH4	OF2	31219	268	60	463	31	107	32	109	36	112	7577	7500	7575	7499	7572	7495
H2	OF2	29952	487	100	1096	56	230	70	244	107	281	7926	7752	7912	7738	7876	7702
N2H4	N2O4	33816	265	73	576	NO INSULATION							4888				
A 50	N2O4	33850	258	71	502	NO INSULATION							4937				
B2H6	OF2	30609	291	64	563	33	110	34	111	36	113	8057	7981	8057	7980	8054	7978
VELOCITY INCREMENT= 35000. FPS																	
H2	O2	36625	673	133	1110	78	316	96	335	145	384	599	760	980	741	931	693
H2	F2	36261	466	78	631	54	216	67	229	100	262	2126	1964	2114	1951	2080	1918
CH4	OF2	37184	301	60	367	34	120	36	122	40	126	1672	1587	1671	1585	1667	1581
H2	OF2	36434	551	99	852	64	260	79	276	119	316	1618	1421	1602	1405	1562	1365
N2H4	N2O4	38475	289	73	530	NO INSULATION							251				
A 50	N2O4	38490	281	71	448	NO INSULATION							329				
B2H6	OF2	36832	328	64	469	37	124	38	125	41	127	1887	1801	1886	1800	1884	1797

* TEN THOUSANDS OF BTUS PER SQUARE FOOT

oxygen combination and its attendant potential payload advantage is compromised by the need for bulky hydrogen tankage leading to increased weight penalties for insulation, expulsion system and structure. The use of the hydrogen-fluorine combination shows the highest potential transport capability in all cases except one with the diborane-oxygen difluoride combination next in rank order.

In interpreting these results it must be noted that they cover cases only where the total heat inleakage through the insulating blanket dominates other sources. In possible cases of interest involving long coast periods and small propellant quantities particularly, this condition may be violated. This limitation has been removed in the evaluation of specific missions and the result can have a marked influence as shown in the succeeding paragraphs.

3. Specific Missions

Tables III through VIII and Tables IX through XIV summarize the results of calculations giving payload estimates for specific missions using non-vented and vented propellant storage methods, respectively. The resulting payload estimates are illustrated in the bar charts of Figures 5 through 10.

We note that, in general, the high energy cryogenic propellants show greater transport capabilities than the earth storable propellants. The use of either the hydrogen-fluorine and diborane-oxygen difluoride combinations results in the greatest payload in every case. The application of the fully mixed, non-vented method of storage results in the greatest payload in all cases except one.

TABLE III

SOLAR ORBIT

NON-VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=35700.FPS TO= 80.DAYS ITO=4649.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PRO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
GROSS WEIGHT= 6000.LBS					M ENG= 73.							
H2	O2	440	4.80	5518	205	21	154	156	213	286	-186	-260
H2	F2	459	12.00	5465	142	12	86	86	168	217	52	2
CH4	OF2	410	5.40	5599	87	9	48	47	23	48	158	134
H2	OF2	450	7.20	5490	166	16	113	115	182	242	-42	-103
N2H4	N2O4	333	1.32	5785	83	11	70	NO	INSULATION		-23	
A 50	N2O4	332	2.06	5787	81	11	59	NO	INSULATION		-13	
B2H6	OF2	429	3.86	5548	96	10	61	60	16	41	195	170
GROSS WEIGHT= 25000.LBS					M ENG= 252.							
H2	O2	440	4.80	22992	503	85	640	636	432	610	94	-80
H2	F2	459	12.00	22771	348	50	350	348	297	418	929	810
CH4	OF2	410	5.40	23331	222	37	213	212	50	113	893	830
H2	OF2	450	7.20	22877	411	63	481	478	350	497	564	420
N2H4	N2O4	333	1.32	24107	212	46	313	NO	INSULATION		68	
A 50	N2O4	332	2.06	24116	207	45	264	NO	INSULATION		114	
B2H6	OF2	429	3.86	23118	243	40	270	269	36	100	1039	976
GROSS WEIGHT= 40000.LBS					M ENG= 379.							
H2	O2	440	4.80	36788	676	133	1075	1070	563	803	383	149
H2	F2	459	12.00	36434	468	78	605	596	379	542	1654	1500
CH4	OF2	410	5.40	37329	301	60	361	358	65	151	1501	1418
H2	OF2	450	7.20	36603	553	99	814	803	452	650	1097	910
N2H4	N2O4	333	1.32	38571	289	73	530	NO	INSULATION		154	
A 50	N2O4	332	2.06	38586	281	71	447	NO	INSULATION		233	
B2H6	OF2	429	3.86	36989	330	64	464	462	47	134	1723	1640

<u>TABLE IV</u>

MERCURY ORBIT

NON-VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=20040.FPS   TO= 90.DAYS   ITO=1841.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PRO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
GROSS WEIGHT= 6000.LBS				M ENG=	73.							
H2	O2	440	4.80	4543	181	21	159	158	90	154	930	867
H2	F2	459	12.00	4455	125	12	94	93	74	118	1164	1121
CH4	OF2	410	5.40	4686	78	9	52	52	13	36	1085	1063
H2	OF2	450	7.20	4497	147	16	124	123	78	130	1064	1012
N2H4	N2O4	333	1.32	5075	76	11	69	NO INSULATION			694	
A 50	N2O4	332	2.06	5081	75	11	59	NO INSULATION			700	
B2H6	OF2	429	3.86	4595	85	10	64	63	11	34	1160	1138
GROSS WEIGHT= 25000.LBS				M ENG=	252.							
H2	O2	440	4.80	18931	445	85	795	790	178	336	4310	4158
H2	F2	459	12.00	18565	306	50	481	479	124	231	5219	5114
CH4	OF2	410	5.40	19528	198	38	266	265	31	87	4685	4629
H2	OF2	450	7.20	18737	362	63	626	623	146	275	4810	4685
N2H4	N2O4	333	1.32	21149	195	46	331	NO INSULATION			3025	
A 50	N2O4	332	2.06	21171	190	45	289	NO INSULATION			3052	
B2H6	OF2	429	3.86	19147	215	40	324	323	27	84	4991	4936
GROSS WEIGHT= 40000.LBS				M ENG=	379.							
H2	O2	440	4.80	30290	598	134	1386	1379	231	443	6979	6774
H2	F2	459	12.00	29704	411	79	838	834	158	301	8428	8288
CH4	OF2	410	5.40	31245	268	60	463	462	41	118	7540	7464
H2	OF2	450	7.20	29980	487	100	1091	1086	187	361	7772	7604
N2H4	N2O4	333	1.32	33839	265	73	576	NO INSULATION			4865	
A 50	N2O4	332	2.06	33873	258	71	502	NO INSULATION			4914	
B2H6	OF2	429	3.86	30636	292	64	564	562	37	113	8026	7950

TABLE V

VENUS ORBIT

NON-VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=11500.FPS TO=108.DAYS ITO=1494.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PRO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
GROSS WEIGHT= 6000.LBS					M ENG= 73.							
H2	O2	440	4.80	3337	149	22	163	162	74	127	2180	2128
H2	F2	459	12.00	3246	102	13	99	99	79	115	2386	2350
CH4	OF2	410	5.40	3491	64	9	55	55	11	29	2295	2277
H2	OF2	450	7.20	3288	120	16	129	128	69	112	2303	2260
N2H4	N2O4	333	1.32	3949	64	11	67	NO	INSULATION		1834	
A 50	N2O4	332	2.06	3955	63	11	59	NO	INSULATION		1836	
B2H6	OF2	429	3.86	3392	69	10	65	65	9	27	2380	2361
GROSS WEIGHT= 25000.LBS					M ENG= 252.							
H2	O2	440	4.80	13905	366	87	821	818	135	265	9431	9304
H2	F2	459	12.00	13526	251	51	504	503	97	184	10317	10230
CH4	OF2	410	5.40	14546	163	38	280	279	25	71	9694	9648
H2	OF2	450	7.20	13703	297	64	651	649	111	218	9919	9814
N2H4	N2O4	333	1.32	16454	165	46	338	NO	INSULATION		7743	
A 50	N2O4	332	2.06	16482	161	45	302	NO	INSULATION		7756	
B2H6	OF2	429	3.86	14134	176	40	330	330	22	69	10042	9996
GROSS WEIGHT= 40000.LBS					M ENG= 379.							
H2	O2	440	4.80	22249	492	136	1427	1423	173	348	15141	14970
H2	F2	459	12.00	21642	337	80	876	875	120	238	16563	16447
CH4	OF2	410	5.40	23273	222	60	486	485	33	96	15544	15481
H2	OF2	450	7.20	21925	400	101	1131	1128	141	284	15920	15779
N2H4	N2O4	333	1.32	26327	225	73	584	NO	INSULATION		12409	
A 50	N2O4	332	2.06	26371	219	71	524	NO	INSULATION		12432	
B2H6	OF2	429	3.86	22615	239	64	573	573	30	93	16096	16034

TABLE VIII

JUPITER ORBIT

NON-VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=40950.FPS TO=610.DAYS ITO=1258.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PRO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
GROSS WEIGHT= 6000.LBS					M ENG=	73.						
H2	O2	440	4.80	5667	208		FUEL STORAGE LIMITS EXCEEDED					
H2	F2	459	12.00	5625	144	FUEL AND OXODIZER STORAGE LIMITS EXCEEDED						
CH4	OF2	410	5.40	5730	89		FUEL STORAGE LIMITS EXCEEDED					
H2	OF2	450	7.20	5645	169		FUEL STORAGE LIMITS EXCEEDED					
N2H4	N2O4	333	1.32	5868	84	11	71	NO INSULATION			-108	
A 50	N2O4	332	2.06	5870	82	11	60	NO INSULATION			-96	
B2H6	OF2	429	3.86	5691	97	9	60	59	13	38	54	29
GROSS WEIGHT= 25000.LBS					M ENG=	252.						
H2	O2	440	4.80	23614	511		FUEL STORAGE LIMITS EXCEEDED					
H2	F2	459	12.00	23438	354		FUEL STORAGE LIMITS EXCEEDED					
CH4	OF2	410	5.40	23879	225	37	206	205	34	98	364	301
H2	OF2	450	7.20	23522	418		FUEL STORAGE LIMITS EXCEEDED					
N2H4	N2O4	333	1.32	24453	214	46	318	NO INSULATION			-284	
A 50	N2O4	332	2.06	24459	209	45	268	NO INSULATION			-234	
B2H6	OF2	429	3.86	23713	247	40	264	263	30	95	451	387
GROSS WEIGHT= 40000.LBS					M ENG=	379.						
H2	O2	440	4.80	37783	687		FUEL STORAGE LIMITS EXCEEDED					
H2	F2	459	12.00	37501	476		FUEL STORAGE LIMITS EXCEEDED					
CH4	OF2	410	5.40	38206	306	60	348	346	44	131	654	568
H2	OF2	450	7.20	37636	563		FUEL STORAGE LIMITS EXCEEDED					
N2H4	N2O4	333	1.32	39125	292	73	538	NO INSULATION			-408	
A 50	N2O4	332	2.06	39135	284	71	453	NO INSULATION			-323	
B2H6	OF2	429	3.86	37942	335	64	446	444	41	129	791	704

TABLE IX

SOLAR ORBIT

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=35700.FPS   TO= 80.DAYS   ITO=4649.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
	GROSS WEIGHT= 6000.LBS				M ENG=		73.						
H2	O2	440	4.80	4056	1588	222	21	225	227	110	189	-297	-379
H2	F2	459	12.00	4100	1497	166	12	142	144	90	150	-83	-144
CH4	OF2	410	5.40	4703	960	83	9	50	50	42	65	77	55
H2	OF2	450	7.20	4208	1400	191	15	183	184	97	167	-170	-241
N2H4	N2O4	333	1.32	5785	0	81	11	70	NO	INSULATION		-21	
A 50	N2O4	332	2.06	5787	0	79	11	59	NO	INSULATION		-10	
B2H6	OF2	429	3.86	4701	916	87	9	62	62	45	67	105	82
	GROSS WEIGHT= 25000.LBS				M ENG=		252.						
H2	O2	440	4.80	19052	4283	519	84	780	777	259	445	-232	-414
H2	F2	459	12.00	18962	4182	365	49	452	449	204	333	531	405
CH4	OF2	410	5.40	20690	2829	213	37	215	214	108	167	653	594
H2	OF2	450	7.20	19493	3697	435	62	612	609	224	380	221	68
N2H4	N2O4	333	1.32	24107	0	210	46	313	NO	INSULATION		70	
A 50	N2O4	332	2.06	24116	0	205	45	264	NO	INSULATION		117	
B2H6	OF2	429	3.86	20639	2681	223	40	268	267	116	174	778	721
	GROSS WEIGHT= 40000.LBS				M ENG=		379.						
H2	O2	440	4.80	31147	6133	695	132	1263	1257	348	596	-100	-342
H2	F2	459	12.00	30863	6116	484	78	719	714	272	442	1085	920
CH4	OF2	410	5.40	33449	4157	290	60	360	359	147	228	1153	1074
H2	OF2	450	7.20	31764	5288	579	98	983	978	300	507	606	403
N2H4	N2O4	333	1.32	38571	0	287	73	530	NO	INSULATION		156	
A 50	N2O4	332	2.06	38586	0	280	71	447	NO	INSULATION		234	
B2H6	OF2	429	3.86	33351	3933	304	64	452	450	159	238	1354	1277

TABLE X

MERCURY ORBIT

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=20040.FPS TO= 90.DAYS ITO=1841.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR		M INS		M PAY	
								A	B	A	B	A	B
	GROSS WEIGHT=	6000.LBS			M ENG=	73.							
H2	O2	440	4.80	4011	703	172	21	178	178	92	154	747	687
H2	F2	459	12.00	3907	737	121	12	112	112	71	114	963	921
CH4	OF2	410	5.40	4297	498	73	9	53	53	41	62	952	932
H2	OF2	450	7.20	4039	610	143	16	144	144	79	130	894	843
N2H4	N2O4	333	1.32	5075	0	74	11	69	NO INSULATION			696	
A 50	N2O4	332	2.06	5081	0	72	11	59	NO INSULATION			702	
B2H6	OF2	429	3.86	4231	474	76	10	64	63	43	63	1026	1006
	GROSS WEIGHT=	25000.LBS			M ENG=	252.							
H2	O2	440	4.80	17437	1973	426	85	831	825	231	382	3763	3618
H2	F2	459	12.00	16967	2151	291	50	511	509	175	276	4600	4501
CH4	OF2	410	5.40	18440	1393	189	38	263	262	106	159	4316	4265
H2	OF2	450	7.20	17498	1653	350	63	669	666	194	319	4318	4197
N2H4	N2O4	333	1.32	21149	0	192	46	331	NO INSULATION			3027	
A 50	N2O4	332	2.06	21171	0	188	45	289	NO INSULATION			3054	
B2H6	OF2	429	3.86	18149	1303	197	40	319	318	111	163	4626	4576
	GROSS WEIGHT=	40000.LBS			M ENG=	379.							
H2	O2	440	4.80	28102	2889	578	133	1433	1425	313	518	6169	5972
H2	F2	459	12.00	27323	3206	392	79	875	872	236	374	7506	7372
CH4	OF2	410	5.40	29641	2054	258	60	457	456	145	217	7003	6932
H2	OF2	450	7.20	28176	2406	473	99	1151	1145	263	432	7049	6886
N2H4	N2O4	333	1.32	33839	0	263	73	576	NO INSULATION			4867	
A 50	N2O4	332	2.06	33873	0	257	71	502	NO INSULATION			4915	
B2H6	OF2	429	3.86	29171	1912	269	64	553	552	152	222	7495	7427

TABLE XI

VENUS ORBIT

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=11500.FPS   TO=108.DAYS   ITO=1494.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
	GROSS WEIGHT= 6000.LBS				M ENG= 73.								
H2	O2	440	4.80	3038	537	141	22	179	178	89	139	1919	1869
H2	F2	459	12.00	2924	595	99	12	118	117	69	104	2107	2073
CH4	OF2	410	5.40	3256	403	61	9	56	56	41	58	2098	2081
H2	OF2	450	7.20	3030	470	117	16	148	147	75	117	2068	2027
N2H4	N2O4	333	1.32	3949	0	62	11	67	NO	INSULATION		1836	
A 50	N2O4	332	2.06	3955	0	61	11	59	NO	INSULATION		1839	
B2H6	OF2	429	3.86	3175	384	63	10	65	65	42	59	2185	2169
	GROSS WEIGHT= 25000.LBS				M ENG= 252.								
H2	O2	440	4.80	13118	1415	348	86	858	855	221	344	8699	8578
H2	F2	459	12.00	12654	1611	237	51	539	538	166	249	9488	9406
CH4	OF2	410	5.40	13955	1015	156	38	280	280	105	149	9195	9152
H2	OF2	450	7.20	13065	1164	284	64	695	693	184	286	9289	9190
N2H4	N2O4	333	1.32	16454	0	163	46	338	NO	INSULATION		7745	
A 50	N2O4	332	2.06	16482	0	159	45	302	NO	INSULATION		7759	
B2H6	OF2	429	3.86	13602	940	162	40	328	328	108	151	9564	9522
	GROSS WEIGHT= 40000.LBS				M ENG= 379.								
H2	O2	440	4.80	21102	2060	472	135	1478	1474	300	467	14069	13906
H2	F2	459	12.00	20354	2379	319	80	921	919	224	336	15340	15230
CH4	OF2	410	5.40	22415	1474	213	60	485	485	144	203	14825	14767
H2	OF2	450	7.20	21008	1673	384	101	1193	1190	250	387	15009	14875
N2H4	N2O4	333	1.32	26327	0	223	73	584	NO	INSULATION		12411	
A 50	N2O4	332	2.06	26371	0	217	71	524	NO	INSULATION		12434	
B2H6	OF2	429	3.86	21848	1355	221	64	569	568	148	206	15412	15355

TABLE XII

LUNAR LANDING

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT= 8850.FPS TO= 3.DAYS ITO= 28.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
	GROSS WEIGHT= 6000.LBS				M ENG=	73.							
H2	O2	440	4.80	2777	24	113	22	159	158	21	61	2808	2768
H2	F2	459	12.00	2695	21	75	13	97	97	15	41	3008	2981
CH4	OF2	410	5.40	2925	15	52	9	53	53	10	25	2860	2845
H2	OF2	450	7.20	2734	20	91	16	126	126	17	50	2919	2887
N2H4	N2O4	333	1.32	3373	0	56	11	64	NO	INSULATION		2420	
A 50	N2O4	332	2.06	3380	0	55	11	58	NO	INSULATION		2422	
B2H6	OF2	429	3.86	2833	14	54	10	62	62	10	25	2940	2926
	GROSS WEIGHT= 25000.LBS				M ENG=	252.							
H2	O2	440	4.80	11592	61	293	87	795	793	56	159	11861	11759
H2	F2	459	12.00	11245	55	195	51	489	489	39	107	12670	12603
CH4	OF2	410	5.40	12201	37	136	38	271	270	27	65	12035	11997
H2	OF2	450	7.20	11411	50	236	65	631	629	45	129	12308	12225
N2H4	N2O4	333	1.32	14056	0	146	46	327	NO	INSULATION		10171	
A 50	N2O4	332	2.06	14083	0	143	45	294	NO	INSULATION		10180	
B2H6	OF2	429	3.86	11817	35	142	41	317	317	27	64	12366	12329
	GROSS WEIGHT= 40000.LBS				M ENG=	379.							
H2	O2	440	4.80	18554	86	400	137	1376	1373	76	218	18989	18850
H2	F2	459	12.00	17997	78	267	81	848	847	53	146	20293	20201
CH4	OF2	410	5.40	19526	52	187	60	470	469	36	89	19286	19234
H2	OF2	450	7.20	18263	69	323	102	1092	1090	62	177	19707	19594
N2H4	N2O4	333	1.32	22490	0	200	73	563	NO	INSULATION		16291	
A 50	N2O4	332	2.06	22533	0	196	72	510	NO	INSULATION		16308	
B2H6	OF2	429	3.86	18911	49	194	65	549	549	38	88	19812	19762

TABLE XIII

MARS ORBIT

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT= 8640.FPS TO=170.DAYS ITO=1217.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
	GROSS WEIGHT= 6000.LBS				M ENG= 73.								
H2	O2	440	4.80	2468	597	128	22	176	176	81	126	2452	2407
H2	F2	459	12.00	2352	689	93	13	121	120	64	96	2593	2561
CH4	OF2	410	5.40	2662	459	55	9	56	56	37	53	2645	2630
H2	OF2	450	7.20	2462	521	107	16	148	148	69	107	2601	2563
N2H4	N2O4	333	1.32	3321	0	56	11	64	NO	INSULATION		2473	
A 50	N2O4	332	2.06	3328	0	54	11	58	NO	INSULATION		2475	
B2H6	OF2	429	3.86	2588	437	57	10	64	64	38	53	2731	2716
	GROSS WEIGHT= 25000.LBS				M ENG= 252.								
H2	O2	440	4.80	10779	1404	309	87	830	828	196	306	11138	11030
H2	F2	459	12.00	10339	1657	212	51	528	527	148	222	11810	11736
CH4	OF2	410	5.40	11546	973	139	38	273	273	94	133	11682	11643
H2	OF2	450	7.20	10740	1103	253	64	676	675	163	254	11745	11656
N2H4	N2O4	333	1.32	13839	0	145	46	325	NO	INSULATION		10390	
A 50	N2O4	332	2.06	13866	0	141	45	293	NO	INSULATION		10400	
B2H6	OF2	429	3.86	11217	891	143	41	315	315	96	133	12042	12005
	GROSS WEIGHT= 40000.LBS				M ENG= 379.								
H2	O2	440	4.80	17356	2009	418	136	1424	1421	266	414	18007	17862
H2	F2	459	12.00	16655	2398	284	80	895	894	199	298	19105	19007
CH4	OF2	410	5.40	18568	1361	189	60	471	471	128	181	18838	18786
H2	OF2	450	7.20	17287	1536	341	101	1154	1153	221	342	18977	18858
N2H4	N2O4	333	1.32	22143	0	198	73	561	NO	INSULATION		16642	
A 50	N2O4	332	2.06	22187	0	194	72	508	NO	INSULATION		16659	
B2H6	OF2	429	3.86	18038	1231	195	65	544	544	131	182	19413	19363

TABLE XIV

JUPITER ORBIT

VENTED PROPELLANT STORAGE

VELOCITY INCREMENT=40950.FPS   TO=610.DAYS   ITO=1258.THOUSAND BTU PER SQUARE FOOT

PROPELLANT		ISP	O/F	M PU	M BO	M TX	M TPA	M STR A	M STR B	M INS A	M INS B	M PAY A	M PAY B
	GROSS WEIGHT= 6000.LBS				M ENG= 73.								
H2	O2	440	4.80	3398	2401	224	21	226	228	62	143	-408	-491
H2	F2	459	12.00	3092	2701	177	12	157	159	53	116	-266	-332
CH4	OF2	410	5.40	4305	1492	86	9	55	55	24	48	-46	-70
H2	OF2	450	7.20	3806	1954	202	15	199	201	57	130	-308	-383
N2H4	N2O4	333	1.32	5868	0	82	11	71	NO INSULATION			-106	
A 50	N2O4	332	2.06	5870	0	79	11	60	NO INSULATION			-94	
B2H6	OF2	429	3.86	4383	1378	88	9	62	62	25	48	-21	-44
	GROSS WEIGHT= 25000.LBS				M ENG= 252.								
H2	O2	440	4.80	18198	5733	521	84	768	774	146	331	-703	-896
H2	F2	459	12.00	17251	6598	367	49	420	424	113	242	-53	-186
CH4	OF2	410	5.40	21262	2739	217	37	215	214	61	121	214	154
H2	OF2	450	7.20	20007	3735	450	62	621	627	129	291	-259	-426
N2H4	N2O4	333	1.32	24453	0	212	46	318	NO INSULATION			-281	
A 50	N2O4	332	2.06	24459	0	207	45	268	NO INSULATION			-232	
B2H6	OF2	429	3.86	21487	2347	225	40	258	257	65	123	324	266
	GROSS WEIGHT= 40000.LBS				M ENG= 379.								
H2	O2	440	4.80	30043	8193	699	132	1229	1239	196	445	-874	-1133
H2	F2	459	12.00	28522	9577	484	78	655	651	150	320	152	-13
CH4	OF2	410	5.40	34737	3631	296	60	358	357	83	165	453	372
H2	OF2	450	7.20	32929	5002	599	98	993	988	172	387	-176	-386
N2H4	N2O4	333	1.32	39125	0	290	73	538	NO INSULATION			-406	
A 50	N2O4	332	2.06	39135	0	283	71	453	NO INSULATION			-322	
B2H6	OF2	429	3.86	35074	3023	306	64	433	431	88	168	630	551

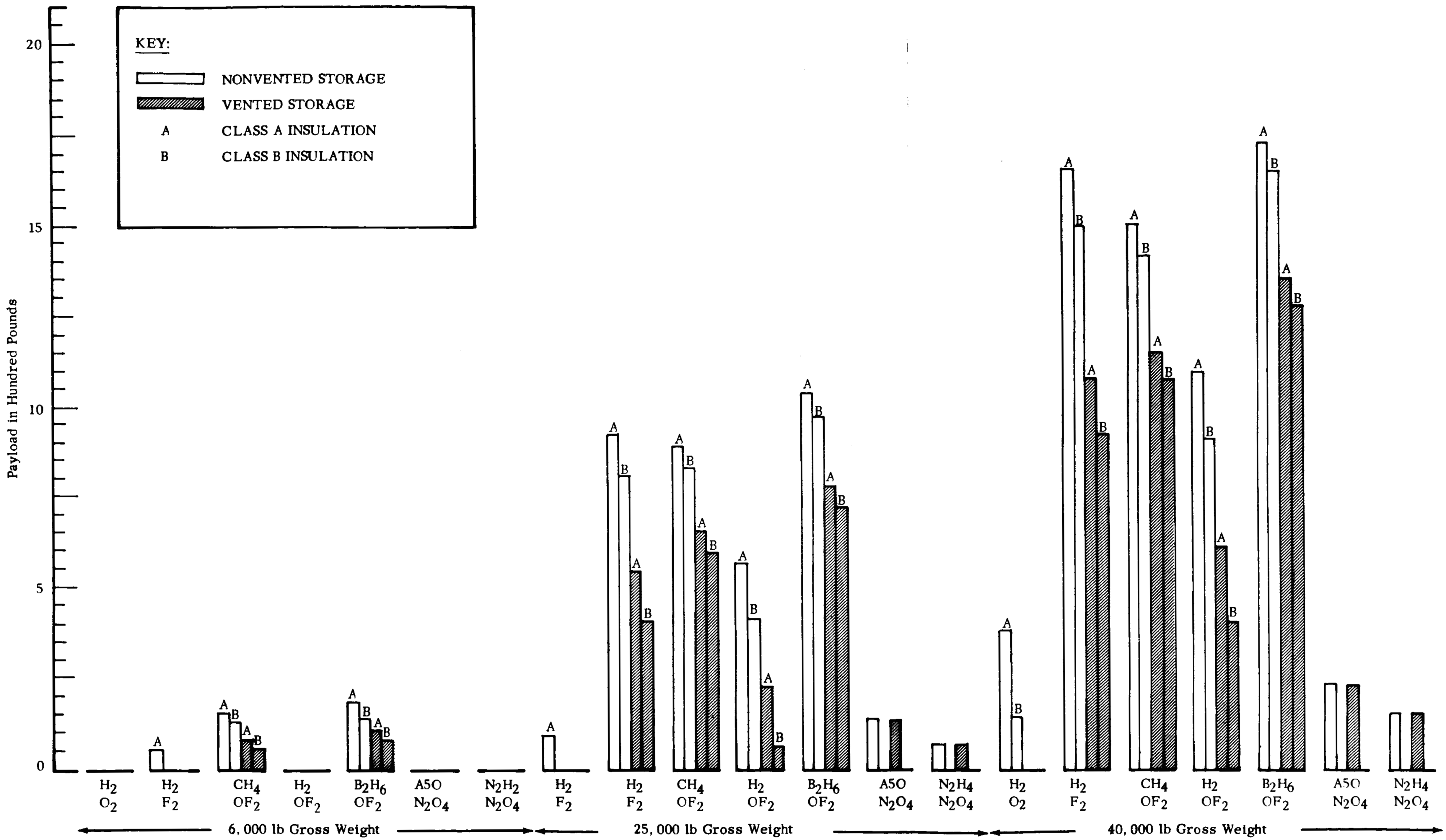

FIGURE 5 PAYLOAD ESTIMATES - SOLAR ORBIT

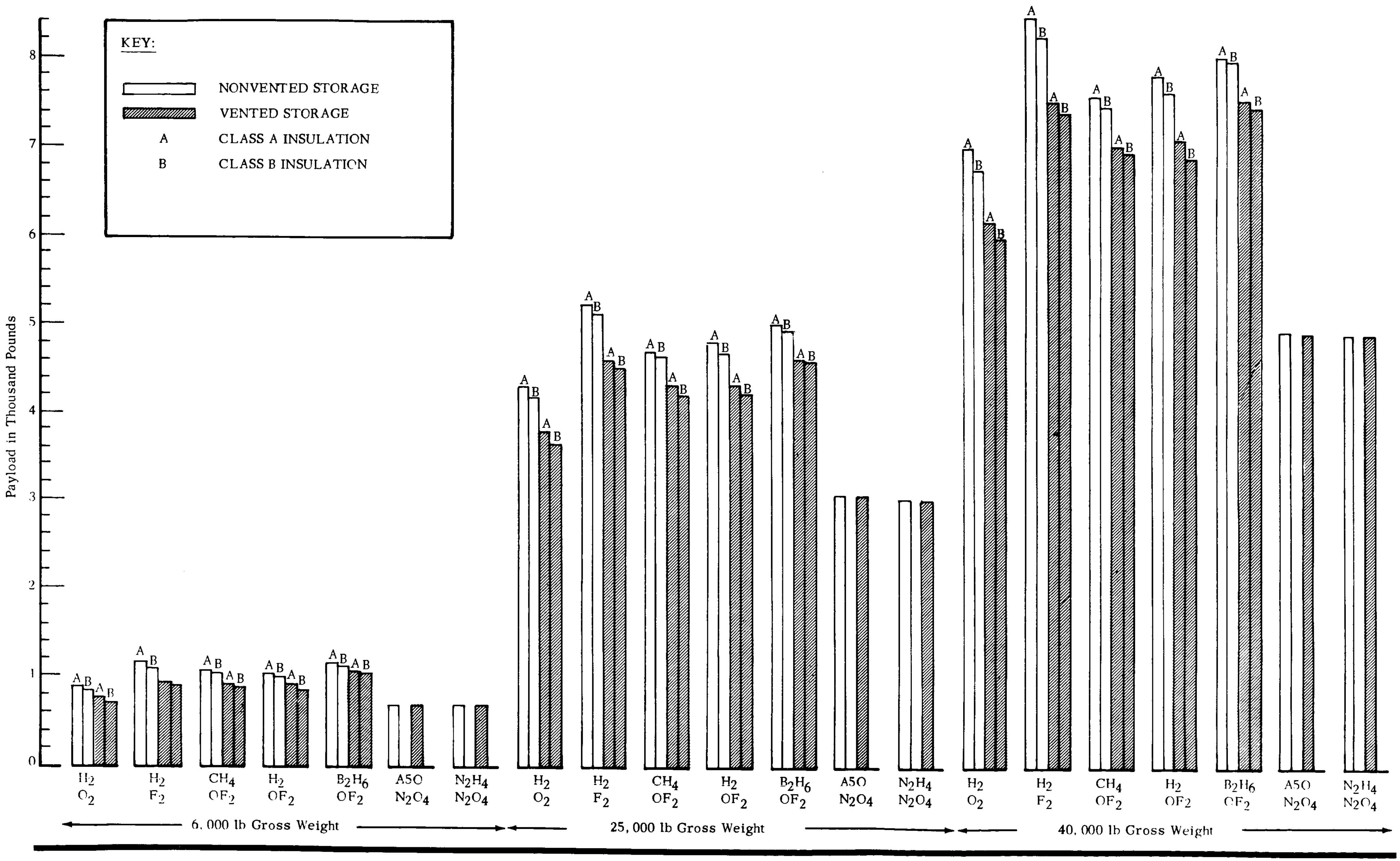

FIGURE 6 PAYLOAD ESTIMATES - MERCURY ORBIT

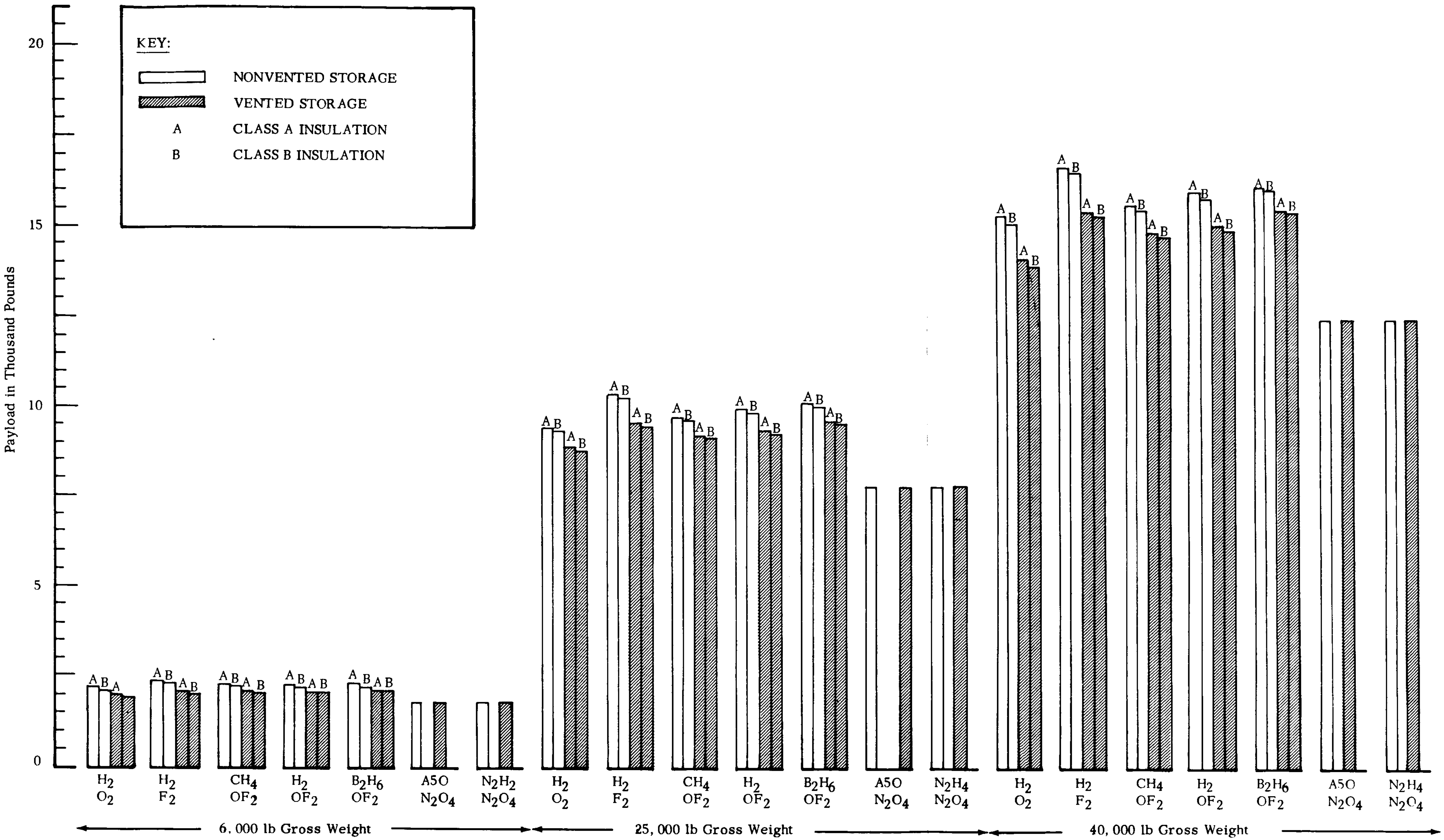

FIGURE 7 PAYLOAD ESTIMATES - VENUS ORBIT

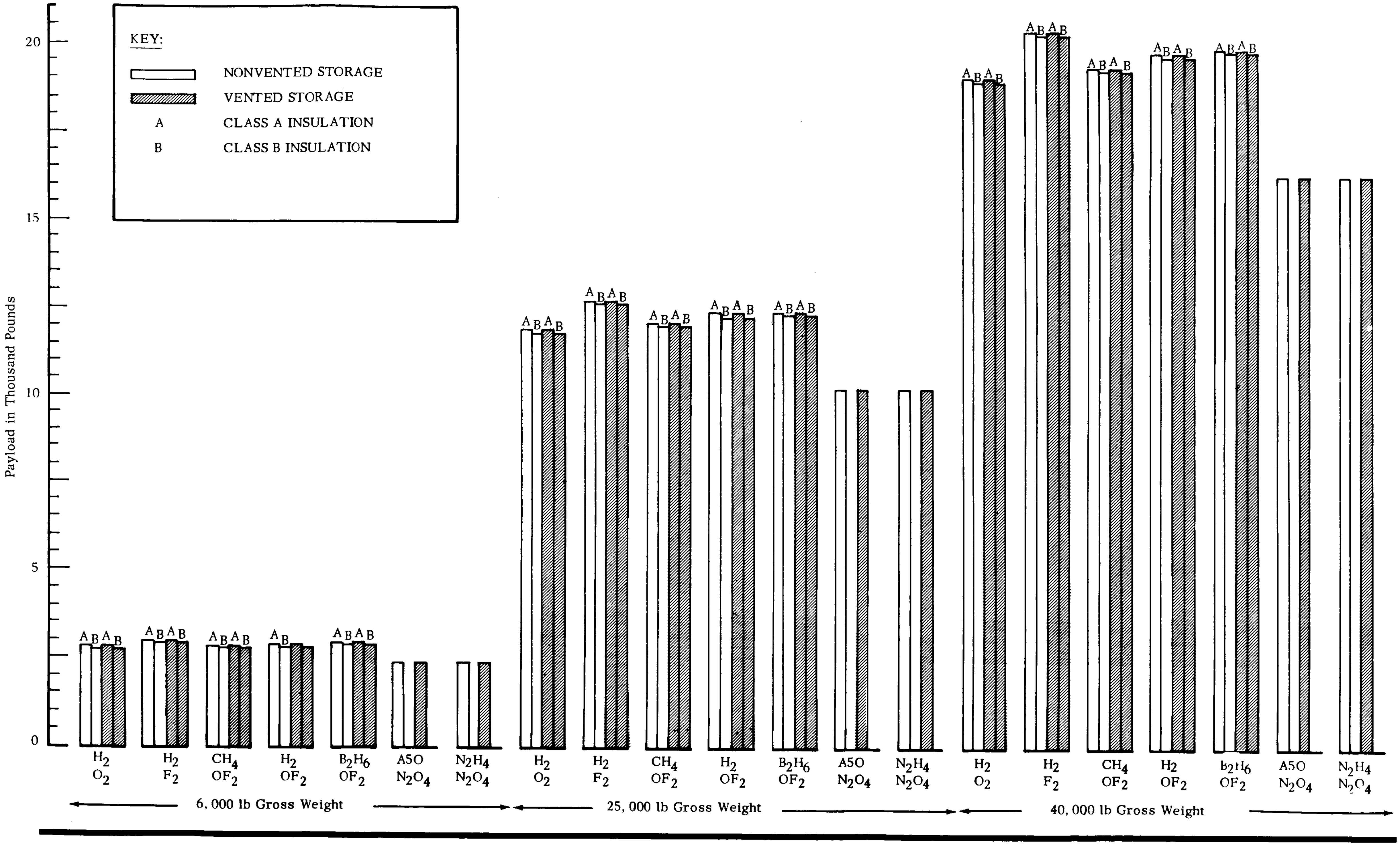

FIGURE 8 PAYLOAD ESTIMATES - LUNAR LANDING

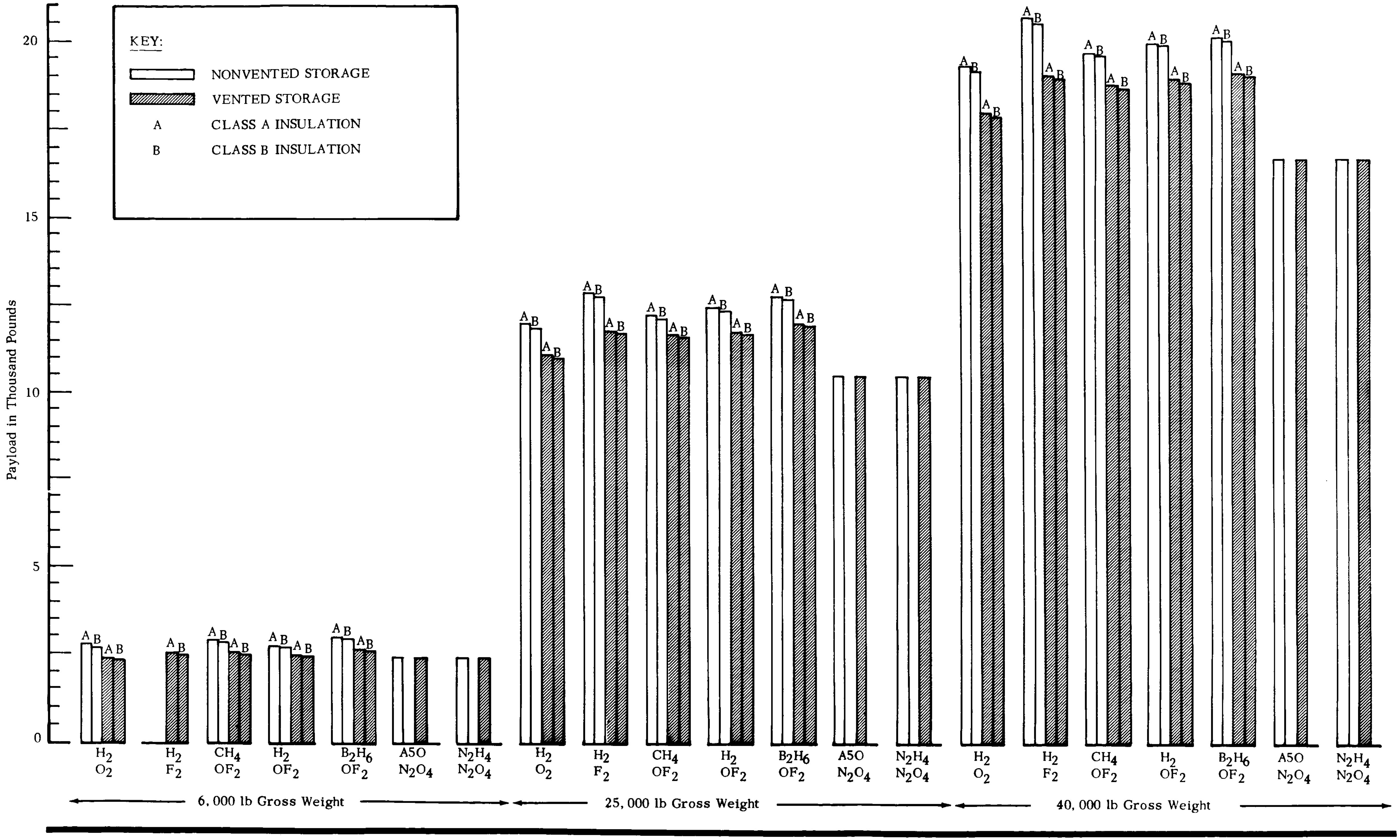

FIGURE 9 PAYLOAD ESTIMATES - MARS ORBIT

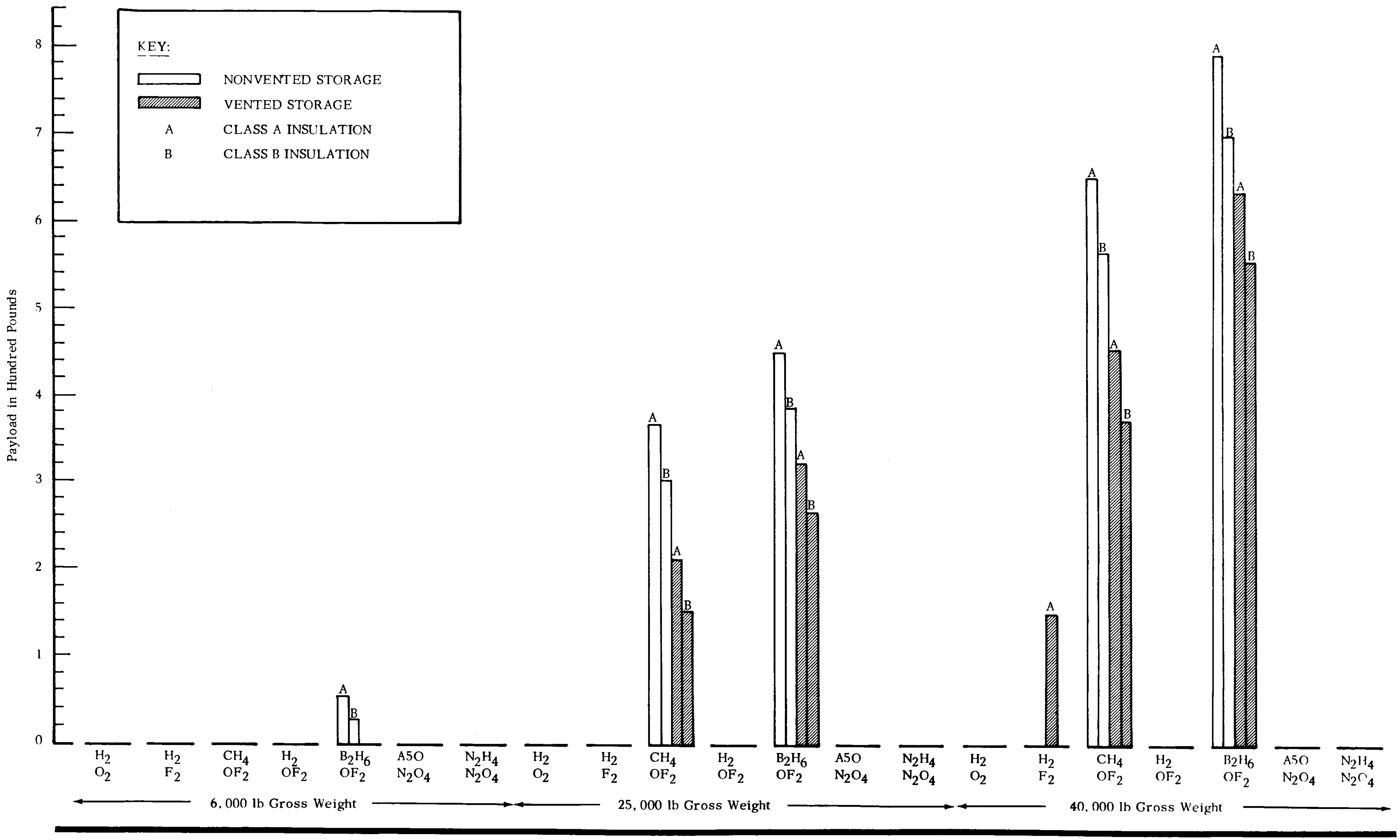

FIGURE 10 PAYLOAD ESTIMATES - JUPITER ORBIT

In a number of cases for the Jupiter mission, heat inleakages via paths not controlled by the insulation blanket lead to pressure increases in the non-vented cryogenic propellant storage containers greater than the limits set. In these cases we might expect to be able to transport measurable payloads (particularly for the larger vehicles) by designing the vehicle expressly for the mission; that is, taking special pains to reduced fixed heat leaks, increasing the pressure capabilities of the storage tanks, etc. Similarly, by tailored vehicle design, the payload potentials resulting from the use of vented storage would be enhanced. Here, we have evidence of the limitations of parametric analyses. Also, we would expect the application of refrigerators to increase the payloads for the Jupiter mission.

Of the cryogenic propellants, the methane-oxygen difluoride and the diborane-oxygen difluoride combinations have physical characteristics which result in compact vehicle design and ease the space storage problem. Their relative advantages in these regards show up in cases where the penalties of space storage are particularly great, for instance, in the Solar and Jupiter missions. The better space storability of diborane (oxygen difluoride) with respect to hydrogen (fluorine) is responsible in those instances where its use shows a greater transport capability.

Finally, we note the relatively poor transport capability of the hydrogen-oxygen combinations compared to the other high energy propellants. The basic reason for this is the relatively large fuel tankage required. This larger tankage requires more structure, more insulation, and a greater weight of expulsion system, all of which subtract from the payload.

4. Special Cases

a. Jettisoning of Part of the Structure

The gain in payload made possible by jettisoning a large fixed fraction (one half) of the structure just before the terminal maneuver, is shown in Table XV, for a solar orbit mission. This mission was chosen because of the high mass ratio u associated with it, and the resulting small payload ratio. It is seen that the increased payload capability is significant, especially for propellants of low density, giving heavy structures.

b. Earth Escape with Partially Filled Tanks

Table XVI shows the effect of using an upper stage vehicle, designed to operate from earth escape, to perform a portion of the earth escape maneuver. This vehicle was designed for a Mars capture, with a gross mass at earth escape of 40,000 lbs., with full H_2/F_2 tanks (see Table XIII).

The independent variable chosen is the amount of propellant, M_P, remaining after the earth escape maneuver. The other three variables shown in Table XVI are functions of M_P. The payload, M_{PL}, of course, decreases as M_P decreases, since the Mars capture maneuver, with H_2/F_2, involves a fixed mass ratio. For the same reason, a non-zero amount of propellant would be required even if the payload were zero.

The actual gross mass, $M_{G(ACTUAL)}$, decreases by the same amount as does M_{PL}.

TABLE XV

GAIN IN PAYLOAD BY JETTISONING PART OF THE STRUCTURE

Mission: Solar Orbit

M_G: 40,000 lb.

ΔV: 35,700 ft/sec.

α: 0.5 (extreme case - half the structure mass jettisoned)

Insulation: Class A (non-vented storage)

Propellant	I_{sp} (sec)	u	$\alpha(1-\frac{1}{u})$	M_{STR}	ΔM_{PL}	Original M_{PL}	New M_{PL}
$H_2 - O_2$	440	12.4	.460	1075	495	383	878
$H_2 - F_2$	459	11.2	.455	605	275	1654	1929
$CH_4 - OF_2$	410	15.0	.466	361	168	1501	1669
$H_2 - OF_2$	450	11.8	.458	814	373	1097	1470
$N_2H_4 - N_2O_4$	333	28.0	.481	530	256	154	410
A50 - N_2O_4	332	28.1	.482	447	216	233	449
$B_2H_6 - OF_2$	429	13.3	.462	464	217	1723	1940

TABLE XVI

PAYLOAD TO MARS, ACTUAL GROSS MASS (FULL TANKS) AND VELOCITY INCREMENT AVAILABLE FOR EARTH ESCAPE VS. PROPELLANT MASS LEFT IN TANKS AT EARTH ESCAPE

Design gross mass:	40,000 lbs.
Propellant:	H_2/F_2
ΔV (Mars Capture):	8,640 ft/sec.

Vented Storage, Class A Insulation

Masses in lb.m.

M_P	M_{PL}	$M_{G(ACTUAL)}$	ΔV_e (ft/sec)
19,053	19,105	40,000	0
15,023	14,038	34,933	1,800
10,023	7,728	28,623	5,600
4,953	1,397	22,292	14,700
3,853	0	20,895	19,200

The velocity increment, ΔV_e, available for the earth escape maneuver, becomes appreciable only when M_{PL} has been greatly reduced. However, even such reduced payloads may be interesting. Finally, for the design gross mass chosen, an upper stage vehicle weighing about 22,000 lbs., could, with some help (an additional ΔV of about 4,000 ft/sec), escape from an earth parking orbit and deliver about 1,000 lbs. of payload to capture around Mars.

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9. Ibid, Figure III-1.

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